2x

NASA

LMSC-D154696

DESIGN GUIDE FOR LOW COST STANDARDIZED PAYLOADS

(NASA-CR-127115) DESIGN GUIDE FOR LOW COST STANDARDIZED PAYLOADS, VOLUME 1 (Lockheed Missiles and Space Co.) 30 Apr. 1972 129 p CSCL 22B
N72-26788
Unclas
G3/31 15746

VOLUME I

30 APRIL 1972

CONTRACT NAS W-2312

LOCKHEED MISSILES & SPACE COMPANY, INC.
A SUBSIDIARY OF LOCKHEED AIRCRAFT CORPORATION

129p

LMSC-D154696
Volume I

DESIGN GUIDE

for

LOW-COST STANDARDIZED PAYLOADS

Volume I

30 April 1972

Contract NAS W-2312

Org. 69-02
Space Systems Division

LOCKHEED MISSILES & SPACE COMPANY

FOREWORD

For the past several years, NASA has been studying the economic merits of new space transportation systems. As part of the most recent economic analyses performed under the direction of NASA/HQ, two Payload Effects studies have contributed strongly to the quantitative validity of payload costs.

As a result of these Payload Effects studies (NAS W-2156 and NAS W-2312) over a period of September 1970 through April 1972, it has been established that savings in payload cost will significantly augment and far exceed the transportation cost savings.

Concept point designs of low-cost and refurbishable spacecraft, subsystems, and modules; and detailed cost estimates thereon; have revealed payload program savings up to 50% of the historical baseline program. Further point designs and costing of standard space hardware; standard subsystem modules, standard spacecraft, and cluster spacecraft; indicated (1) further savings are possible and (2) specific operational advantages are attainable with the Shuttle that were hitherto unattainable with conventional launch vehicles.

Now that the Shuttle has been determined to be economically desirable, it is necessary to implement the low-cost payload designs not only to obtain the economic benefits, but to insure that proper interfaces with the Shuttle system are planned and designed in parallel with the Shuttle development.

Space program managers and planners and payload designers now have the opportunity to initiate the "Payload Revolution", and thus (1) expand the scope and accomplishments of their programs and/or (2) assure the retention of their programs under the future austere funding constraints by offering lower-cost payloads to accomplish the required missions.

This document, "Design Guide for Low-Cost Standardized Payloads", presents guidelines for the design of low-cost and refurbishable payloads, and defines a methodology for the design of standardized space hardware. The concepts presented may appear to some to be revolutionary and controversial; however, because of the potential benefits of their application, they deserve the careful consideration of all who are vitally interested in the vigorous continuation of space programs in the late 1970's and beyond.

TABLE OF CONTENTS

DESIGN GUIDE FOR LOW-COST STANDARDIZED PAYLOADS

TABLE OF CONTENTS (CONT.)

TABLE OF CONTENTS (CONT.)

TABLE OF CONTENTS (CONT.)

DEFINITION OF TERMS

Many of the terms used herein have various connotations within the Aerospace community. Therefore, as a guide, some of the basic terms are defined below.

PAYLOAD SYSTEM	describes collectively: (1) the payload; (2) the payload/Shuttle adapters, and any deployment or separation devices required to effect a separation of the payload from the launch vehicle; (3) payload ground support equipment; (4) payload flight support equipment including spare module support racks, payload checkout equipment, and special payload umbilicals.
PAYLOAD	the total operating entity, such as a satellite, that is launched into orbit by the Shuttle; it comprises spacecraft and experiments but excludes Shuttle related elements - such as platforms or adapters - that are non-functional relevant to the orbiting satellite.
BASELINE PAYLOAD	a current unmanned payload used to provide a basis for the development of low-cost or standard payloads and for cost comparisons.
LOW-COST PAYLOAD	A payload designed for launch by the Space Shuttle or by a future large-expendable launch vehicle. Such a payload is designed (1) without the traditional costly constraints on weight and volume, and (2) for in-orbit repair or refurbishment.
SUBSYSTEM	A major functional group of equipment which is essential to the operation of a spacecraft. Spacecraft subsystems include: • Structures & Mechanisms • Electrical Power • Stabilization & Control • Attitude Control • Communications, Data Processing & Instrumentation • Environmental Control • Propulsion

COMPONENT	an assembly such as a star tracker, transmitter, or similar. Components are assemblies of parts.
PART	a piece of hardware, a quantity of which are assembled into a single component; examples are: transistor, lens, shaft, etc.
STANDARD SUBSYSTEM	a major spacecraft subsystem (stabilization and control; communication, data processing, and instrumentation; electrical power; attitude control) designed for application to a significant number of mission-peculiar or standard spacecraft.
STANDARD SUBSYSTEM MODULE	a plug-in assembly of components forming a major segment of a standard subsystem, and having standard mechanical, electrical, and thermal interfaces.
STANDARD SUBSYSTEM MODULE VARIANT	a standard subsystem module modified by the addition, deletion, or substitution of a single component.
STANDARD SPACECRAFT	a small quantity of different types of spacecraft incorporating standard subsystem modules, each type capable of replacing a significant number of the mission-peculiar spacecraft defined in the NASA mission model. The spaceframe, integral wiring harnesses, and thermal control elements of each standard spacecraft type are standardized.
CLUSTER SPACECRAFT	a spacecraft incorporating standard subsystem modules and capable of supporting concurrently the experiment/sensor packages of several of the missions defined by the NASA Mission Model.
RELIABILITY	the probability that a system, subsystem, component, or part will satisfactorily perform its intended function without catastrophic failure for a prescribed period of time, within a prescribed environment.
CONFIDENCE LEVEL	the probability that the reliability figure-of-merit predicted for a system, subsystem, component, or part is correct.
REPAIR	an action taken to restore a failed system to an operating state. The action may be scheduled or unscheduled, and consists of: • Diagnosis of the failure condition • Removal of the failed system element • Replacement of the failed element with a similar element in operating condition • Checkout of the system post-maintenance to assure proper operation within prescribed limits.

PARTIAL REFURBISHMENT	A maintenance action expected to prevent future failure. In this study it is assumed that when a repair visit to the system becomes necessary to repair a failed system element, other system elements which have not yet failed will be approaching their theoretical point of first failure. These latter elements will be removed also and replaced as assurance that the system will be protected against failures occurring soon after a repair visit.
FULL REFURBISHMENT	A maintenance action (analogical to a complete overhaul) occurring shortly prior to, or at the theoretical MMD point of the system, where MMD denotes the useful operating life terminal point as dictated by the limits of the design. The action consists of removal and replacement of all dynamic system elements, whether or not they have exhibited failure. Following full refurbishment, the spacecraft is assumed to be in the "as new" state and capable of operating another period equal to the spacecraft MMD.
MAINTENANCE LEVEL (SPACECRAFT)	The hardware level at which maintenance action takes place. Since the systems in question are modularized at the subsystem level, all maintenance actions are confined to removal and replacement of the module, or modules exhibiting failure, or approaching a theoretical failure point.
MAINTENANCE INTERVAL	That period of elapsed time between any one maintenance action and the next, as scheduled in the overall maintenance program. The interval is predicated upon expectable wearout rates, and expected failure incidence.

Section 1
INTRODUCTION

This document, "Design Guide for Low-Cost Standardized Payloads*", has been prepared by Lockheed Missiles & Space Company, Inc., Space Systems Division, as part of the total effort under NASA Contract NAS W-2312, covering the Payload Effects Follow-On Study. The specific effort for the Design Guide was identified as Task 1.9. The following paragraphs provide a brief background of the Payload Effects studies and reference associated reports.

1.1 The Overall Objective

NASA/HQ, in its implementation of the Payload Effects studies, established the basic objectives of designing some typical future payloads and determining their operational and cost impacts upon the Space Program in the Space Shuttle era, beginning in 1979.

1.2 Initial Payload Effects Study and Design Guide

The first Payload Effects study was completed by LMSC in June, 1971. The results were documented in a Final Report, LMSC-A990556, dtd 30 June 1971.

Three historical payloads were redesigned to:

a. Incorporate low-cost features, including many resulting from relaxation of design weight and volume constraints (use of Space Shuttle or new-large-expendable launch vehicles provides significantly increased weight and volume capability for payloads)

* The word "Payload" as used in this document, refers to a flight vehicle, usually comprising a spacecraft and an experiment/sensor package, which is carried to orbit in the cargo bay of the Space Shuttle. For additional definitions, consult the "Definition of Terms".

b. Incorporate modular equipment packaging to allow on-orbit replacement (for repair, refurbishment, or update) of spacecraft or experiment modules either by an astronaut in EVA or by remotely-controlled automated devices such as manipulators or teleoperators.

The three low-cost refurbishable payloads for which new designs were created were:

Payload	Cognizant Agency
• Orbiting Astronomical Observatory (OAO)	NASA/Goddard (Grumman)
• Synchronous Equatorial Orbiter (SEO)*	NASA/Langley (Boeing)
• Small Research Satellite	USAF (LMSC)

The derivation of these designs, the program costs therefore, and the economic impact analyses are described in the Final Report, LMSC-A990556.

The low-cost design approaches applied are described in detail in the document, LMSC-A990558, dtd 30 June 1971, "Design Guide for Space Shuttle Low-Cost Payloads".

1.3 Follow-On Payload Effects Study

The results of the initial study effort indicated a dramatic impact of payloads upon the 1979 and beyond space program. In fact, it appeared that unless payload savings could be implemented (by use of low-cost payload design/manufacturing/testing techniques and by refurbishment/reuse of payloads), the Shuttle program would not be economically feasible. For this reason, a follow-on study was sponsored by NASA/HQ, with co-direction from two directorates, OSSA and OMSF.

---

* Modified design using Lunar Orbiter baseline design.

The new study had the following principal objectives:

- Create additional point designs of future spacecraft, incorporating not only features of low-cost and refurbishability, but also establishing spacecraft hardware standardization at three levels*:

  (1) Standard Subsystems and Modules
  (2) Standard Spacecraft
  (3) Cluster Spacecraft

- Prepare program plans and cost estimates for the low-cost standardized spacecraft hardware and establish dollar savings related to baseline (traditional design) payload programs.

- Determine effect of the new payload designs upon the Space Shuttle system and the constraints which the Shuttle (in its current configuration) might place upon full realization of future payload cost reduction potential.

- Prepare a Designers Guide (as a sequel to the initial document mentioned in para. 1.2), updating the principles of low-cost payload design for Space Shuttle application and providing additional methodology for design and application of standard hardware to future spacecraft.

1.4 The Updated Design Guide

This document is the result of effort on the last item listed above. It summarizes the economic impact of low-cost, standardized spacecraft and provides special information pertinent to implementation of the critically-needed payload cost-reduction principles.

The data has been prepared for use not alone by project engineers and designers, but also by mission planners, program managers, and all others who bear the responsibility for implementing the required cost-effective payload programs of the future.

---

* Definitions are provided elsewhere in the document.

Section 2
CONTENT AND USAGE OF GUIDE

2.1 Content of Design Guide

The objective of this document is to communicate to space program managers, mission planners, designers, engineers, and scientists:

(1) Basic relationships between future payload designs and program cost effects (Section 3)

(2) Special impact of mission commonalization and space hardware standardization (Section 4)

(3) Methods for optimizing spacecraft design life, reliability, repair/refurbishment (Section 5)

(4) Specific concepts for designing low-cost, refurbishable, and standard spacecraft and subsystems (Sections 6 and 7)

2.2 Application to Pre-Shuttle Payloads

Although emphasis has been placed upon application of special design techniques for Shuttle-carried payloads, most of the principles presented can also be applied with success to near-future expendable and booster-launched spacecraft.

a. Tradeoffs indicate clearly that many of the larger and/or more costly spacecraft, when designed with low-cost principles (and therefore larger and heavier), can be launched on a larger and more expensive launch vehicle with a significant net dollar saving (lower payload cost outweighs increased launch vehicle cost).

b. Application of low-density module approach to spacecraft equipment packaging will significantly reduce bench assembly, final assembly, and testing costs and allow rapid removal/replacement for field repair prior to launch.

2.3 Application to Shuttle Payloads

The total NASA/USAF Mission Model, which lists all the Space Shuttle traffic for the 1979-1990 time period, includes several types of payloads which will be carried by the Shuttle. Principal categories are:

a. Unmanned Programs

- Earth Satellites
- Planetary Orbiters
- Space Probes
- Propulsion Stages or Tugs

b. Manned Programs

- Capsules/Platforms for Shuttle-Sortie Missions
- Logistic Supply Capsules for Man Support
- Large Experiment Modules
- Space Station Segments

The payload principles discussed herein were developed primarily for unmanned satellites and planetary orbiters. However, the low-cost and standardization premises presented are applicable in a general sense to all unmanned payloads and to certain of the manned-program payloads.

2.4 Emphasis on General Principles

Although quantitative data is provided, this document is not intended as a design handbook to instruct designers in detail hardware design and analysis. Rather, it provides general guides by which cost-effective payloads can be derived and implemented.

2.5 Division of Data into Two Volumes

To allow inclusion of a considerable amount of future payload design reference data without burdening the casual reader/user with many pages of detail, the Design Guide has been divided into two volumes:

Volume I - Basic Design Guide

Volume II - A collection of Engineering Memos describing the detail point designs of low-cost standard spacecraft and subsystems/modules thereof.

Figure 2-1 is a tabulation of these LMSC memos.

Type Hardware	Application	S/C General Description	Stabilization & Control	Communications Data Processing & Instrumentat.	Electrical Power	Attitude Control	Propulsion
Standard Subsystems	EOS COMSAT Planetary	PE-106 PE-126 -	PE-102 PE-122 -	PE-103 PE-123 PE-133	PE-104 PE-124 -	PE-105 PE-125 -	- - PE-137
Standard Spacecraft	EOS LAOS COMSAT	PE-156 PE-146 PE-126	- - -	- - -	- - -	- - -	- - -
Cluster Spacecraft	CLEOS CLAOS	PE-166 PE-186	- -	- -	- -	- -	- -

EOS = Earth Observatory Satellite
COMSAT = Communication Satellite
LAOS = Large Astronomical Observatory Spacecraft
CLEOS = Cluster Earth Observatory Spacecraft
CLAOS = Cluster Astronomical Observatory Spacecraft

Fig. 2-1 Tabulation of Engineering Memos

Section 3
GENERAL RELATIONSHIP OF PAYLOAD APPROACHES TO PROGRAM COSTS

3.1 Typical Program Cost Breakdown

It is important to identify all payload-related program costs to insure that none are omitted when cost-effectiveness tradeoffs are performed in the planning of a space program. The principal cost categories are:

- Research Development Test and Evaluation (RDT&E)
- Unit Recurring (also referred to as Investment)
- Operations (a recurring cost)

These cost categories are broken down into sub-elements in Fig. 3-1.

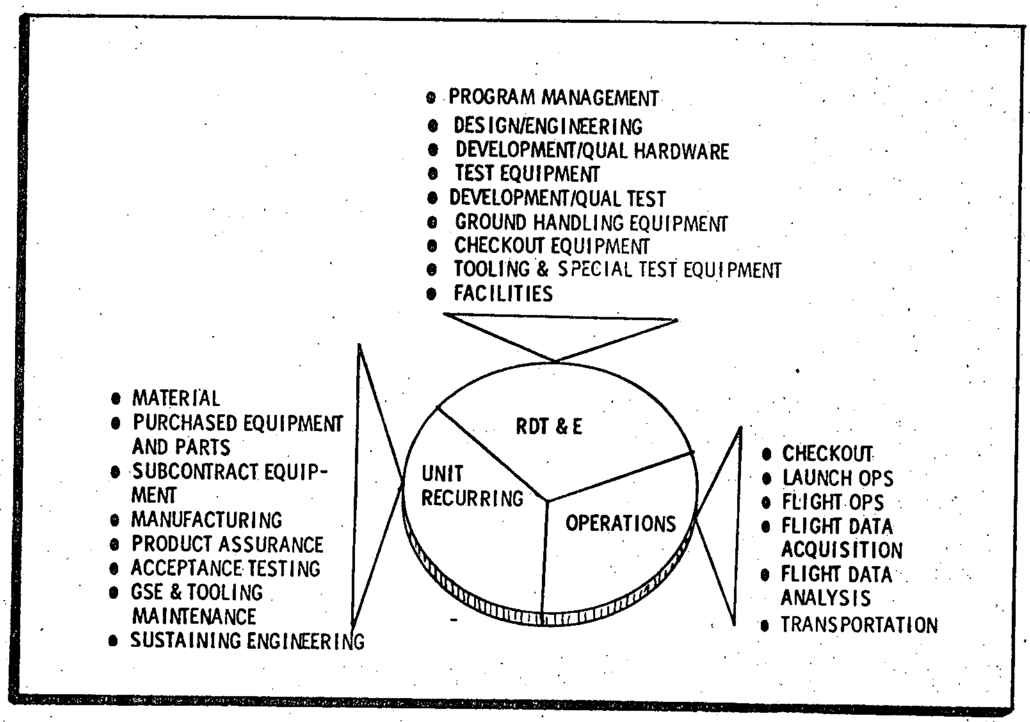


Fig. 3-1 Breakdown of Space Program Costs

3.2 Relative Effect of RDT&E and Recurring Cost Savings

3.2.1 Discounted $ Effects

In making cost tradeoffs to determine which of alternative designs to select, situations occur in which lower costs may be obtained in either RDT&E or in recurring costs (or in both). If there is an option between otherwise essentially equal choices, it is usually better to select the approach which results in higher dollar saving in RDT&E. Most comparative economic analyses of funding and benefits for long term projects are now based upon a discounting principle wherein an allowance is made for the time-dependency of the value of money. Utilizing compound interest principles, benefits derived several years in the future must be discounted to the same period of time that the expenditures are incurred in order to evaluate the costs and the benefits by the same "yardstick". Thus, savings in the RDT&E spending usually precede other expenditures by several years. However, if the time interval between the expenditure and the benefit is small (such as might be the case for high-volume assembly line techniques), the inverse might be true.

3.2.2 Influence of Unit Recurring Cost on RDT&E Costs

For typical development and qualification test programs, even at the reduced-level planned for low-cost payloads, up to 1.5 equivalent payloads (varies depending on complexity of payload) are used as test hardware. These costs are part of the RDT&E total for testing. This fact indicates that multiple savings can be accomplished by reduction of payload unit recurring cost; a saving in both unit cost and RDT&E will result. The aforementioned discounting (par. 3.2.1) provides additional savings in the portion of cost allocatable to RDT&E.

3.3 Impact of Payload Design on Program Costs

Payload design at all levels; system, subsystem, component, or part; significantly affects all categories of program costs. Payload managers and designers must consider the cost impact of all design decisions if program costs are to be minimized. This requires an awareness of the cost consequences of these decisions; an awareness that will arise only through study of the design-cost relationships.

There are a large number of potential payload cost-reduction areas which come into being as a result of using the Space Shuttle. These have been reduced to a reasonably brief list as shown on Fig. 3-2 and are discussed in subsequent sections. The "X" entered in each block indicates to which cost categories each cost reduction approach is applicable. This basic form, or a similar one, can be used as an initial checklist to determine if all cost-reduction possibilities are being pursued in the design of a particular payload.

Cost Reduction Area	RDT&E									Unit Production					Operations			
	Prog. Mgmt.	P/L Integr.	Development Des. Engr.	Development Qual. Hdwe.	Development Qual. Test	Test Equip.	Tooling	Checkout Equipment	Facilities	Matl./Parts	Purch. Equipment	Manufactur.	Prod. Assur.	Accep. Test	Checkout	Launch Ops	Flight Ops	Flt. Data Acq/Anal.
Simplified Contract/Document Requirements	X	X	X		X					X	X	X	X	X				
Simplified Configuration Management (Traceability)	X	X	X		X					X	X	X	X	X				
Use Proven Technology-Off Shelf Hdwe.			X		X									X				X
Use Low-Cost Materials				X	X		X			X		X	X	X				
Decrease Stress Level on Parts				X	X					X	X			X	X			
Use Lower-Quality Parts			X	X	X		X		X	X	X	X	X	X	X			
Increase Structure Safety Margin			X	X	X	X	X		X	X	X	X	X	X	X			
User Lower Reliability Goals (On-Orbit Maintenance)			X	X	X	X		X		X	X		X	X	X			
Increase Volume of Packages			X				X				X	X	X					
Increase Hardware Weight Allowance			X	X			X			X	X	X	X					
Simplify/Modularize Hardware			X	X	X	X		X		X	X	X	X	X	X			X
Simplify/Reduce Ground Testing	X	X	X	X	X	X			X									
Use Orbit Maintenance/Refurb. & Reuse	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Employ Standard Hardware	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Use Pre-Deployment Orbit Checkout			X	X	X	X		X	X	X	X		X	X	X		X	X

Fig. 3-2 Potential Cost Reduction Areas

Specific approaches to and guidelines for low-cost payload system design, together with descriptions of the impact upon costs, are provided in Section 6 of this Design Guide.

Similar data on the cost impact of spacecraft hardware standardization upon program costs are provided in Section 7.

3.4 General Impact on Refurbishment and Reuse on Payload Program Cost

Section 5 of the Design Guide provides considerable discussion on the effects of payload reliability, payload design life, and degree of repair/refurbishment (and combinations thereof) upon payload program costs. The following paragraphs are provided as an over-view, pointing out the significant cost-impact features of typical payload programs employing refurbishment/reuse in lieu of expendable payloads.

3.4.1 Design of Refurbishable Hardware

As explained in par. 6.2.9, spacecraft hardware must be modularized to allow for in-orbit replacement of equipment modules. Typical examples of these modules and the accepting spacecraft are described in paragraphs 7.1.5 and 7.4.2. The subassemblies and components within the modules must also be designed to allow ease of refurbishment in field depots when the used/failed modules are returned to earth by the Shuttle.

Careful design of the spacecraft, modules, components, and parts for ease of refurbishment will add very little, if any, to the design and development costs for a spacecraft.

The production costs (unit recurring) for the refurbishable hardware will be lower for the following reasons:

- Simple bench assembly and testing of components and modules can be accomplished
- Subsystems can be pre-tested prior to spaceframe installation by interconnecting a group of modules
- Components or parts can be readily inspected or replaced if malfunction occurs
- Final assembly of subsystems into spaceframe is grossly simplified (modules on slide-in rails).

In summary, it is desirable from both a cost and operational standpoint to modularize the spacecraft equipment, even on expendable payloads. The only minor penalty for incorporating full refurbishability is the addition of support rails and latches (in lieu of bolt-down fittings) to attach the modules into the spaceframe.

3.4.2 Benefits of On-Orbit Spacecraft Refurbishment

Once the spacecraft modules have been designed for on-orbit replacement, two specific benefits accrue:

3.4.2.1 Repair of Launch/Ascent Failures. A significant percentage of all historical spacecraft failures have occurred in the launch/ascent phase (this is discussed in detail in par. 5.2.4). During the initial payload placement, the Shuttle can carry a set of spare spacecraft modules to orbit with the payload. A checkout of the payload prior to orbit deployment (using on-board checkout set) will reveal any failures which have occurred during the rigorous launch/ascent phase. The defective modules can be replaced using on-board Teleoperator or manipulators (or EVA as backup), the checkout can be repeated and the payload deployed. This basic approach will save millions of dollars by essentially reducing to zero payload losses resulting from launch/ascent and early-operation in orbit.

3.4.2.2 Cycling of Modules in Lieu of Payloads. One of the concepts of payload refurbishment involves delivery of a replacement payload to orbit and retrieving and returning the used/failed payload to earth for refurbishment. This approach provides the maximum-possible workload upon the Shuttle and Space Tug.

An improved concept developed by LMSC provides for payload refurbishment to be accomplished in orbit and only the modules to be returned to earth for refurbishment/reuse cycling. This module-cycling approach is particularly beneficial in refurbishment of payloads in Syneq orbit because of the limited capability of the Space Tug to deliver and retrieve complete payloads. It also allows multiple-payload refurbishment in low-earth orbit on single Shuttle flights.

The cost savings resulting from earth-orbit-earth module-cycling in lieu of payload-cycling are quite significant, representing large multiples of the $10.5 million per Shuttle flight. The quantitative cost savings data presented in

Section 4 do not include the full cost benefits of multi-mission refurbishments; however, the effects of refurbishment in orbit of both Syneq orbit and LEO payloads have been included.

Performance analyses to date indicate that refurbishment of multiple payloads is feasible:

- A single Shuttle flight carrying replacement modules can revisit and refurbish three or more payloads in a low-earth common orbit. (It is not feasible to deliver the replacement payloads to orbit and return the spent payloads to earth with a single Shuttle flight).
- A pair of Shuttle flights combined with a Tandem Tug launch from LEO can deliver replacement modules and perform refurbishment in Syneq orbit of two or more payloads, thus averaging one Shuttle/Tug flight per payload refurbishment. (A single Shuttle/Tug cannot deliver to Syneq orbit and return to earth a typical future payload.)

3.4.3 Benefits of Ground Refurbishment

The used/failed modules returned from orbit by the Shuttle can be processed through a ground refurbishment cycle and restored to their initial or "new" function and life expectancy. The cost savings resulting from this hardware cycling can be very significant.

In actual design analyses performed on typical payload subsystem modules, the ratio of refurbishment cost to the cost of a new replacement module has varied from as low as 9% to as high as 60%; a weighted average is about 20%. This means that there is a potential of 80% average cost saving in the payload hardware cost for each refurbishment. The transportation cost for placing a new payload or revisiting the orbiting payload for orbit-refurbishment could be approximately the same. However, the cost advantage is biased toward the refurbishment because (as mentioned previously) round-tripping a set of replacement modules to orbit costs less on the average than round-tripping the total payload.

The summary results of cost analyses for 45 NASA plus non-NASA missions for the 1979-1990 time period are shown in Fig. 3-3. Of the total payload savings of

$5.987 billion, refurbishment/reuse of payload modules accounted for $2.377 billion. The other part of the savings was the result of low-cost design approaches used in the payload programs.

Cost Element	$ Million				
	Baseline Expendable Payloads	Low-Cost Refurbished Payloads	Savings		
			Total	Low-Cost	Re-furb
Payload	$ 16169 M	$ 10182 M	$ 5987	$ 3610	$ 2377
Transportation	3608	4404	(-796)	(-398)	(-399)
Total Program	$ 19777 M	$ 14586 M	$ 5191	$ 3212	$ 1978

Fig. 3-3 Cost Comparison - Low-Cost/Refurbished Payloads

3.5 General Impact of Standardized Hardware on Payload Program Costs

The cost impact of standardized hardware has been analyzed at three different levels of implementation:

a. Standard Subsystems - applied to mission-peculiar spacecraft

b. Multi-Mission Standard Spacecraft - replacing single mission-peculiar spacecraft

c. Cluster Spacecraft - replacing groups of mission-peculiar spacecraft

Section 7 includes a detailed presentation of the standard hardware concepts and examples of typical design approaches. Section 4 provides a description of the types and amounts of cost reduction attainable with standard hardware. The following paragraphs provide a brief over-view of the cost impacts.

3.5.1 Standard Subsystems

Standard Subsystems have been applied to all missions of the NASA Mission Model except missions beyond the orbit of Mars.

The standardization of this large and variable amount of spacecraft hardware will result in very important cost savings, primarily in the RDT&E phase of programs. Because RDT&E costs of standard hardware may be shared by many programs, the burden of RDT&E costs is reduced for each program and in total. Also, the Government will realize significant savings in overall space program operations costs as a result of reducing the total variety and inventory of spacecraft hardware.

Historically, it was believed that the provision of a limited inventory of standard subsystems may lead to "overkill" of spacecraft requirements; for example, the substitution of a standard subsystem may provide capability in excess of that provided by a mission-peculiar subsystem, and thus increase the unit recurring cost of the spacecraft. To minimize this potential cost penalty, subsystems have been standardized at the module level (two or more modules per subsystem). Variants of basic modules may also be obtained by the simple addition, deletion, or substitution of standard plug-in components within the module.

In addition, analyses have shown that the savings to be realized through the sharing of subsystem RDT&E costs greatly exceed the minor cost penalties of "overkill". In summary, the net savings due to standardization of subsystems are significant and should be pursued.

3.5.2 Standard Spacecraft

Additional cost-savings may be realized by the use of a small number of different Standard Spacecraft to perform most of the space missions. Such Standard Spacecraft incorporate Standard Subsystems. Major cost savings can be realized in spacecraft-level RDT&E (spaceframe, integral wiring, etc.) systems integration, and operations costs; as compared to the costs of performing the same missions with totally program-peculiar spacecraft.

3.5.3 Cluster Spacecraft

Cluster Spacecraft, in particular one designed to perform earth-observation missions in low-earth orbits, can reduce the number of spacecraft required in orbit, thus reducing the total costs of spacecraft procurement, transportation and in-orbit maintenance.

Section 4

COST REDUCTIONS AFFORDED BY LOW-COST STANDARDIZED PAYLOADS

In addition to those payload program cost reductions available through implementation of low-cost payload designs and refurbishment/reuse of payload hardware (discussed in Section 3), there is a large array of further cost savings made possible by standardization of future payload program elements.

Because of the common objective, a cost-effective space program, principles of standardization should not be implemented without inclusion of the corollary low-cost, refurbishable payload characteristics. This section of the Design Guide therefore addresses composite cost reductions, with reference occasionally to those attributable to standardization alone.

Two different areas of cost reductions are discussed in this section:

(1) Cost Savings Resulting from "Standardized" Missions
(2) Cost Savings Potential in Standard Space Hardware

The data presented will offer a brief explanation of the nature of the cost reduction, its derivation, and typical quantitative cost data to illustrate the magnitude of the potential savings. The cost data provided in this section has been derived from the separate LMSC report "Cost Impact of Low-Cost Standard Space Hardware" dtd 29 February 1972.

4.1 Cost Reduction by Mission Commonalization

One of the initial steps in applying space program standardization is to evaluate the potential of combining mutually-compatible missions. A total step process might include:

a. Consolidation of Orbits
b. Combination of Mission Experiments on Multi-Purpose Spacecraft

Implementation of these approaches would result in space transportation savings. The following paragraphs offer specific rationale for the cost reduction.

4.1.1 Consolidation of Orbits

4.1.1.1 Low-Earth Orbit Missions. In separate analyses, it has been demonstrated that many low-earth orbit (LEO) missions listed in the NASA mission model currently planned for various altitudes and inclinations, can be consolidated into two basic groups and three specific orbits:

(1) 600 km, 30° inclination

(2) 500 km, 97.4° inclination sun-synchronous

a. Dawn-Dusk
b. Noon-Midnight

The candidate missions for consolidation are listed on Fig. 4-1. Some of the missions cannot be relocated to the common orbits and are listed as "not combinable."

4.1.1.2 Syneq Orbit Missions. These missions are, by description, in a common orbit. In fact, because of preferred ground targets and ground stations, the separate missions will probably be competing for specific preferred points in the orbit plane.

4.1.2 Experiment Combinations on Multi-Purpose Spacecraft

4.1.2.1 LEO Missions. Although commonalization of mission orbits offers some direct transportation cost savings (explained in par. 4.1.3), the next logical step can be made. This involves the combination of experiments from two or more missions upon a multi-purpose spacecraft. The only requirement is that experiment packages so combined be compatible relative to target pointing. Tangible examples of the extreme of this approach are described in sub-section 7.5.5 as "Cluster Spacecraft." The payload cost savings accruing with use of the Cluster Spacecraft are provided in par. 4.2.3 following.

Comb.	Composite Mission			Baseline Missions			
	Alt. (KM)	Incl.	Orbit Time	Fleming No.	Mission Name	Alt. (KM)	Incl.
A	600	30°	-	1	Astronomy Explorer A	500	28.5°
				6	OSO	650	30°
				13	HEAO-C	650	28.5°
				15	LST	650	28.5°
				17	LSO	650	30°
				19	LRO	650	30°
B	500	97.4° SS	Noon/ Mid-night	21	Polar EOS	930	99° SS
				25	TIROS	1300	101° SS
				26	Polar ERS	930	99° SS
				75	TOS Met.	1300	101° SS
				77	Polar ERS	930	99° SS
C	500	97.4° SS	Dawn/ Dusk	26	Polar ERS	930	99° SS
				77	Polar ERS	930	99° SS
-	Not Combinable			3	Magnetosphere - Low	3520/ 260	28.5°- 90°
				7	Gravity/Relativity	930	90°
				23	Earth Physics	740	90°
				30	Small ATS	5550/ 555	90°
				32	Co-op ATS		

Fig. 4-1 Candidates for LEO Mission Combinations

4.1.2.2 Syneq Orbit Missions. The combination of two or more mission-experiment sets upon a single Syneq spacecraft is perhaps more attractive than for LEO missions. The combination would eliminate some of the aforementioned competition for desirable points in the orbit. Also, because most of the Syneq missions require earth-pointing, a single spacecraft can more readily support a multi-mission assignment in Syneq orbit.

Although no point designs for a Syneq Cluster Spacecraft have been developed, the concept is totally feasible.

4.1.3 Transportation Sharing

The principal cost-saving resulting from mission commonalization accrues from the reduction of space transportation costs. The three areas from which these lower costs will be derived are discussed in the following paragraphs.

4.1.3.1 Multi-Payload Placement. With payloads planned for placement into a common orbit, it is possible to schedule the Shuttle launches so that two or more payloads can be placed using a single Shuttle flight. This will directly reduce the Shuttle usage cost allocated to each mission (one Shuttle flight costs $7.3 million; a Space Tug flight costs $0.6 million additional).

a. LEO Placement

Analyses have indicated that sufficient maneuvering capability exists in the Shuttle to place up to four payloads at different points in a common low-earth orbit.

b. Syneq Orbit Placement

Although the placement of multiple payloads into Syneq orbit is feasible, the combined capability of the Shuttle and Space Tug provides some constraints. The use of tandem Tugs, however, provides sufficient capability for placement of at least two of the large future payloads.

4.1.3.2 Multi-Mission Revisit. In the revisit mode, the common orbit also provides savings. The Shuttle's ability to revisit two or more payloads in different orbits on a single flight is quite limited. The maximum plane change capability, for example is on the order of 4 degrees. However, with payloads at different points in the same orbit, up to four payloads can be revisited using a single Shuttle flight.

4.1.3.3 Cluster Placement and Revisit. The placement of a Cluster in low-earth orbit can be accomplished using one or two Shuttle flights (typical Cluster configurations shown in par. 7.5.5). The equivalent mission-equipment mounted on several mission-peculiar spacecraft would require additional Shuttle delivery flights.

In the revisit mode, the Shuttle can carry more cargo (spare spacecraft modules, experiment update equipment, etc.) to a single Cluster rendezvous point than to several different points even in the same orbit plane. The Cluster, therefore, provides the ultimate requirement for lowest-cost transportation.

4.1.4 Approximate Cost Reductions

As a single program cost influence, the commonalization of missions can provide transportation savings approaching $1 billion for the total 1979-1990 space program (out of an initial total of approximately $4 billion for recurring Shuttle costs).

The increment of transportation saving provided by use of the LEO Cluster in lieu of individual mission-peculiar spacecraft is about $280 million.

4.2 Cost Reduction by Standardization of Space Payloads

As a separate means of obtaining space program cost reduction, the various elements of the payload hardware can be standardized. The description of approaches used and detail of typical designs of this low-cost standardized hardware is provided in Section 6 (low-cost spacecraft design) and in Section 7 (Standard spacecraft design). The following cost reduction data are provided in three segments:

- Standard Subsystems and Modules
- Standard Spaceccaft
- Cluster Spacecraft

4.2.1 Standard Subsystems and Modules

4.2.1.1 Cost Advantages. The first major step in development of standard hardware approach consists of standardizing the subsystems of the collection of mission-peculiar spacecraft and then further refining them into universal-application standard modules. This process is thoroughly described in subsection 7.1. The following is a listing of some of the cost advantages accruing from this standardization:

a. Reduction of design and development costs - one design will serve several spacecraft rather than each requiring peculiar subsystem designs.

b. Reduction in procurement costs - parts and components may be purchased in larger quantities with resultant savings to programs.

c. Reduction in quantity and types of specifications required.

d. Reduction in testing - a standardized set of tests can be prepared for all spacecraft using the same module types.

e. Reduction in manufacturing costs - fabrication facilities can be scheduled to make all modules of the same kind in one production run, rather than fabricate one-of-a-kind hardware items. As a secondary benefit, the reliability and quality of the hardware can also be improved because longer production runs facilitate "debugging" to eliminate production anomalies, and workmanship skills improve with familiarity with a given set of production operations.

f. Reduction in the costs of quality control - it is demonstrable that more homogeneity exists in a process containing larger numbers of similar hardware items; process anomalies and correction will be fewer.

g. Reduction in assembly time - the benefit of large batch production is realized in consonance with standard learning curves.

h. Reduction in the costs of logistics and spare parts -

- Lead time for delivery is reduced if items are standard rather than one-of-a-kind peculiars.
- Standard modules permit ordering of batch lots of replacement parts and components, with purchase cost advantages and inventory control simplification.
- All modules for a given program of spacecraft can be delivered to using facilities, and stocked on a first-in/first-out basis.

i. Reduction in ground/flight operations costs - use of fewer hardware elements for total space program will decrease training and field maintenance costs, reduce launch operations support, and simplify data acquisition and reduction.

j. Simplification of Test Equipment and Facilities - standardization of test equipment and facilities, allows similar logistic and spares benefits in their regard, and simplifies the design requirements imposed upon such test equipment.

4.2.1.2 <u>Cost Reductions with Standard Modules</u>. The program costs which accrue from the application of standard subsystem modules to mission-peculiar spacecraft for 45 NASA plus non-NASA missions (LEO, Syneq orbit, high-energy orbit, and planetary missions) are tabulated in Fig. 4-2. The baseline payloads are fully program-peculiar, of traditional design, and are launched in an expendable mode.

Cost Category	Cost ($ Million)*		
	Baseline Expendable Payload	Low-Cost Spacecraft Standard Subsystems	Savings
RDT&E	$ 7038M	$ 3917M	$3121M
Unit (Investment)	8104	4288	3816
Operations	1027	1253	(-226)
Payload Total	$16169M	$ 9458M	$6711M

* For space program 1979-1990.

Fig. 4-2 Cost Comparison-Standard Subsystem Modules
(45 NASA + Non-NASA Missions)

The savings in payload RDT&E costs alone are over $3. billion or 45% of the baseline cost; the overall payload program savings are $6.7 billion.

The baseline payloads are launched by "current expendable" launch vehicles; the total transportation cost for the baseline case is $3.6 billion versus an estimated $4.4 billion for the Shuttle/Tug support of the spacecraft with low-cost, refurbishable spacecraft incorporating standard modules. A large portion of the Shuttle/Tug costs result from an increased quantity of flights to support the cost-saving payload refurbishment and reuse program.

The standard subsystem savings, extrapolated to the total 1979-1990 space program, account for total payload program savings (exclusive of transportation costs) of approximately $12 billion.

The aforementioned savings included the composite effects of low-cost design, spacecraft refurbishment/reuse, and standard subsystems. The isolated effect of standard subsystem application alone is, in RDT&E costs: over $.7 billion savings for the selected 45 missions and $1.6 billion for the total space program.

4.2.2 Standard Spacecraft

4.2.2.1 Cost Advantages. Supplementing the development and implementation of standard subsystems and modules, the remaining mission-peculiar hardware elements of the spacecraft can be standardized; these include the spaceframe, special mechanisms, integral wiring, and thermal control elements. The development of the Standard Spacecraft concept and typical point designs are described in sub-section 7.4.

All of the cost advantages accruing to spacecraft as a result of using standard modules also apply to the Standard Spacecraft. In addition, there are other cost advantages to be derived from using a Standard Spacecraft for a group of missions. Some major cost advantages are listed:

a. RDT&E cost reduction - The separate design and development of several different spacecraft can be replaced by a single design and development program. Although potentially more costly than any of the mission-peculiar spacecraft developments it replaces, the amortized cost per mission for the Standard Spacecraft is far less costly.

b. Space logistics cost reduction - the spare modules to be carried to orbit by the Shuttle for repair/refurbishment of spacecraft can be identical for all missions supported by a single type of Standard Spacecraft, simplifying the hardware logistics. The total quantity of each module carried (for multi-mission revisit) can also be fewer.

c. Total procurement spans can be reduced - all spacecraft required for a set of missions can be procured under a single procurement schedule, thereby reducing need for standby or sustaining teams (as required for intermittent procurement of several mission-peculiar spacecraft). Delivered spacecraft can be stored for future use and monitored periodically for operational readiness as necessary (similar to Polaris, Saturn and Minuteman programs).

d. Payload support equipment cost reduction - a significant simplification in ground handling, servicing, and checkout equipment can be implemented with attendant cost savings.

e. Field operations cost reductions - simplification in field personnel training and reduction of ground maintenance costs will result from the use of a very few different basic spacecraft.

f. Shuttle interface equipment standardization - the implementation of a small quantity of Standard Spacecraft will allow simplification and standardization of Shuttle interface equipment and ground/flight operating techniques; smaller Shuttle support crews and reduced quantities of ground station personnel will be required.

4.2.2.2 Cost Reductions with Standard Spacecraft. The payload program costs accruing from application of low-cost Standard Spacecraft to replace mission-peculiar spacecraft for 15 low-earth orbit missions are tabulated in Fig. 4-3, and compared to (1) the expendable payload baseline costs, and (2) the equivalent 15 missions with low-cost standard subsystems.

Compared to the baseline, the Standard Spacecraft (for these 15 missions) will provide a cost saving of $2.25 billion in RDT&E, $2.26 billion in Unit costs and an overall payload program saving of $4.5 billion. Most of these savings accrue from use of the standard subsystems, however. The increment of additional savings accountable to the Standard Spacecraft are in the RDT&E category and equal $400 million.

Cost Category	Cost ($ Million)*			
	Baseline Expendable Payloads	Low-Cost Standard Subsystems	Low-Cost Standard Spacecraft	Savings/with Standard S/C
RDT&E	$ 3649 M	$ 1792 M	$ 1392 M	$ 2257 M
Unit (Investment)	4017	1756	1756	2261
Operations	548	506	506	42
Payload Total	$ 8214 M	$ 4054 M	$ 3654 M	$ 4560 M

* for space program 1979-1990

Fig. 4-3 Cost Comparison - Standard Spacecraft
(15 LEO NASA + Non-NASA Missions)

There is little difference in transportation cost between the standard subsystem case and the Standard Spacecraft case; respectively, the recurring Shuttle operations costs are $1.16 billion and $ .98 billion (compared to the baseline Transportation costs of $ .89 billion).

Although Standard Spacecraft for missions other than low-earth orbit are feasible and desirable, specific cost-savings have not as yet been estimated. The savings would be principally in spacecraft RDT&E; a very rough extrapolation for the total 1979-1990 space program indicates that savings approaching $1 billion could be obtained in addition to those obtainable with use of standard subsystems.

4.2.3 Cluster Spacecraft

4.2.3.1 Cost Advantages. The use of Cluster payloads in lieu of individual mission-peculiar or Standard Spacecraft is the final step in standardization of space hardware. The development of the Cluster Spacecraft concept and typical point designs are described in sub-section 7.5.

The basic cost advantages accruing to spacecraft, as a result of (1) using standard subsystem modules and (2) using Standard Spacecraft, also apply to Cluster Spacecraft. In addition, there are other cost advantages to be gained from use of Cluster Spacecraft to replace groups of separate payloads. The major additional cost advantages are:

a. RDT&E cost reduction - because all engineering and development is concentrated on development of one spacecraft in lieu of several different types, cost savings will result. The cost of Cluster development will be more than that for any mission-peculiar spacecraft, but the cost apportionment to several missions will reduce the per-mission RDT&E.

b. Hardware investment cost reduction - a lesser variety and quantity of subsystem modules will be required for the Cluster spacecraft than for equivalent Standard Spacecraft.

c. Transportation cost reduction - a fewer quantity of Shuttle flights will be required to provide repair and refurbishment to the Cluster Spacecraft and to provide revisit to and updating of experiment packages.

d. Space logistics cost-reduction - the limited types of subsystem modules on the Cluster will allow consolidation and reduction of spares required for repair.

e. Ground network cost reduction - ground-link communications from a very few Cluster payloads will result in consolidation and standardization of data transmission, collection, and processing methods with attendant savings in personnel and facilities.

f. Payload support cost reduction - replacement of several different payloads with a single Cluster will require a minimum of different types of GSE and will reduce ground crew training and personnel count.

g. Shuttle interface equipment cost reduction - the various equipment used to support, deploy, and retrieve payloads can be consolidated into only one or two universal-application configurations.

4.3.2.2 Cost Reduction with Cluster Spacecraft. The payload program costs accruing from application of low-cost Cluster Spacecraft to replace mission-peculiar spacecraft for 11 low-earth orbit missions are tabulated in Fig. 4-4, and compared to (1) payload baseline costs and (2) the equivalent missions with low-cost standard subsystems. Four of the 15 missions used for cost comparisons in par. 4.2.2.2 are not applicable to the LEO Clusters.

Cost Category	Cost ($ Million)*			
	Baseline Expendable Payloads	Low-Cost Standard Subsystems	Low-Cost Cluster Spacecraft	Savings/w Cluster S/C
RDT&E Unit (Investment) Operations	$ 3133M 3738 504	$ 1417M 1543 453	$ 1178M 1326 485	$ 1955M 2412 19
Payload Subtotal	$ 7375M	$ 3413M	$ 2989M	$ 4386M
Transportation	744	985	540	204
Total Program	$ 8119M	$ 4398M	$ 3529M	$ 4590M

* for space program 1979-1990

Fig. 4-4 Cost Comparison-Cluster Spacecraft
(11 LEO NASA + Non-NASA Missions)

Compared to the baseline, the Cluster Spacecraft (for 11 missions) will provide a cost saving of almost $2 billion in RDT&E, $2.4 billion in Unit costs, and an overall payload program saving of $4.6 billion.

Most of the savings accrue from use of standard subsystems. The increments of additional savings allocatable to Cluster Spacecraft alone are:

Cluster Spacecraft RDT&E	$ 239 million
Cluster Spacecraft Investment	217 "
Cluster Transportation	869 "
Cluster Operations	(-32) "
Total	$1293 million

4.2.4 Best-Mix of Standard Hardware

4.2.4.1 Selection of Hardware. To obtain the optimum low-cost space program, it is desirable to establish a "best-mix" of standard space hardware. A typical listing of 45 NASA plus non-NASA missions, with selection of standard hardware approach, is shown on Fig. 4-5. The mix is comprised of 30 missions with standard subsystems, 4 with standard spacecraft, and 11 with Cluster Spacecraft.

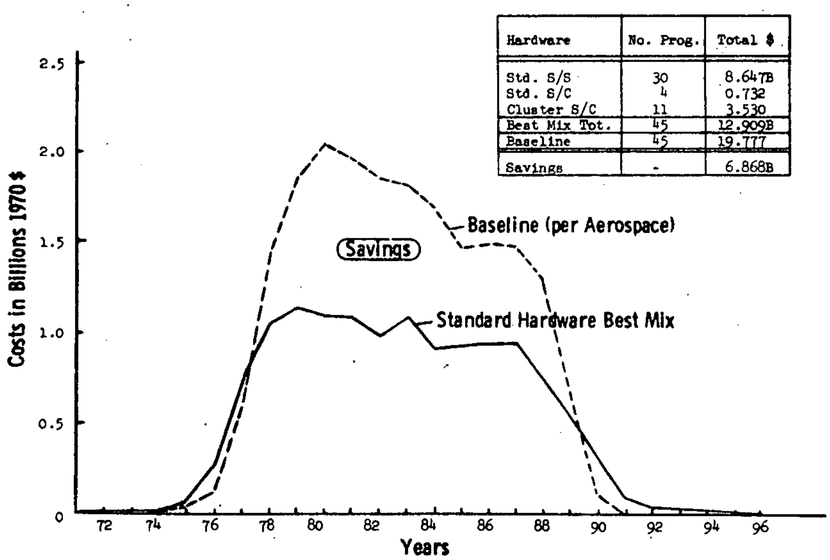
4.2.4.2 Cost Reduction for Best-Mix. The cost reduction and cost spreads for the typical best-mix standard hardware applied to 45 missions is shown on Fig. 4-6.

It is apparent that the peak funding level (1980) can be reduced by approximately $1 billion; the average annual funding is reduced from about $1.75 billion per year to about $1 billion per year. The total savings for the 45 missions is almost $6.9 billion for the 1979-1990 time span.

The costs shown in the inset on Fig. 4-6 include Transportation costs. The equivalent total for the standard subsystems case is $13.862 billion. The best-mix affords an additional $953 savings of which $453 million are transportation (Shuttle recurring operations) savings.

	Mission	Selected Approach: Standard Subsystem	Standard Spacecraft	Cluster Spacecraft
1	Astronomy Explorer			#1
2	Astronomy Explorer B	X		
3	Magnetosphere-Low	X		
4	Magnetosphere-Middle	X		
5	Magnetosphere-Upper	X		
6	OSO			#1
7	Gravity Relativity		#1	
8	Gravity Relativity B&D	X		
9	Radio Interferometer	X		
10	Solar Orbit Pair-A	X		
11	Solar Orbit Pair-B	X		
12	Optical Interferometer	X		
13	HEAO			#1
15	LGT			#1
17	LSO			#1
19	LRO			#1
21	Polar EOS			#2
22	SEO	X		
23	Earth Physics		1	
24	Sync. Met.	X		
25	TIROS			#2
26	Polar ERS			#2
27	Sync. ER	X		
28	ATS	X		
29	Small ATS-B	X		
30	Small ATS-A		#1	
31	Coop ATS-A	X		
32	Coop ATS-B		#1	
33	Medical Network	X		
34	Educat. Broadcast	X		
35	Follow-On Sys. Demonst.	X		
36	TDRS	X		
50	Mars Viking	X		
52	Venus Explorer-Orbiter	X		
53	Venus Radar Mapping	X		
54	Venus Explorer Lander	X		
70	COMSATs	X		
71	US Domestic Commun.	X		
72	Foreign Domestic Commun.	X		
73	NAV/Traffic Control B	X		
74	NAV/Traffic Control A	X		
75	TIROS Op. Met.			#2
76	Sync. Met.	X		
77	Polar ER			#2
78	Sync. ER	X		

Fig. 4-5 Best-Mix of Standard Hardware
(45 NASA + Non-NASA Missions)



Hardware	No. Prog.	Total $
Std. S/S	30	8.647B
Std. S/C	4	0.732
Cluster S/C	11	3.530
Best Mix Tot.	45	12.909B
Baseline	45	19.777
Savings	-	6.868B

Fig. 4-6 Cost Breakdown and Spread
for Best-Mix Standardization

The example and costs discussed in the foregoing represents only a portion of the total space program missions. The application of Standard Spacecraft and/ or Cluster Spacecraft to the other missions certainly is feasible.

The best-mix to be derived from a larger or different assemblage of space missions will be different from the one shown here. There is one general conclusion, however: there are literally billions of dollars of cost reduction which can be obtained by application of low-cost, refurbishable/reusable, and standardized spacecraft.

Section 5
COST-EFFECTIVE APPLICATION OF PAYLOAD RELIABILITY, MMD, REPAIR, REFURBISHMENT

To the program manager, the project engineer, the senior space systems designer, and others concerned with the provision of the maximum space payload performance for the least expenditure of funds, the relationships existing among reliability, maintainability, mean mission duration (MMD), confidence level, and costs are of considerable importance.

For a space program to achieve some given level of success, a series of compromise decisions must be made in order that the mission success required may be achieved in a cost effective manner. This section considers the means by which reliability, maintenance, MMD, and confidence levels may be optimized with respect to program costs, and describes the subordinate factors contributing to the relationships among the variables. All of these major variables have significant separate and combined effects upon payload RDT&E, payload unit, and on-orbit operations costs; therefore, each variable is discussed separately in terms of cost optimization, as well as in terms of its inter-relationships with the other variables involved.

5.1 Cost Impact of Payload Reliability and Confidence Level

The implementation of the Space Shuttle will make possible a new generation of payloads for which no precedent exists. These payloads will be maintainable, whereas prior US unmanned payloads have been inaccessible and hence unmaintainable after launch. Such payloads have been termed "Expendable", and once the first major orbital failure occurs, the payload has little or no further utility.

The historical expendable payloads have been designed to obviate failure as far as possible, and where failure is anticipated, each potentially failing component is offset by another identical component using the techniques of redundancy.

The penalty paid for redundancy is additional weight and system complexity, as well as additional cost. In many cases, additional electrical power to support the additional componentry also is needed. These weight-increase influences, coupled with the limited weight and volume capability of the conventional expendable booster vehicles, forced the designer to use high-density equipment packaging approaches. Because the densely packaged small units tended to fail more rapidly than larger, less densely packaged hardware of similar function, the booster weight and volume constraints resulted in costly national programs to improve the reliability of unit parts.

The cost-aggravating historical constraints will change markedly when the Space Shuttle becomes available as the launch/service transportation vehicle for space payloads; for example:

- Space hardware will no longer be so severely weight- and volume-limited
- Designs will not be required to be secured against any, and all, failures
- Space hardware liftimes without failure can be foreshortened and the costs of high reliability can be avoided
- Spacecraft can be modularized for ease of in-space service, repair, refurbishment
- Standard modules can be designed for use on a series of spacecraft so that module replacement costs can be reduced in the overall spacecraft maintenance plan.

5.1.1 Reliability Values and Confidence Levels

5.1.1.1 Initial Assignment of R and CL Values. Assignment of a reliability goal and confidence level to a space program has been the responsibility of the program manager, and will continue to be so for space programs employing maintainable spacecraft. Customarily, the space program manager issues a requirement for one or more space vehicles for which the chance of failure is the minimum possible within the program budget and the dictates of the state-of-the-art. He also requires the strongest possible guarantee that what is asked for is within the inherent design capability of the vehicle. The aerospace contractor, in responding to the requirement, has sought to (a) predict that the reliability of the final vehicle will be as specified, (b) to afford assurance that the prediction is valid, and (c) to attach a price which is not only competitive but also fairly represents what the program manager can afford. Axiomatically, the higher the reliability requirement and its associated confidence level are, the higher the cost of the system must be.

5.1.1.2 Definitions. To discuss the interactions of reliability values and confidence levels requires the use of statistical mathematics, which can become quite abstruse; and it is not the intention of this section to do so. However, for those to whom the terms reliability and confidence level may be unfamiliar, the following definitions are essential to the understanding of their cost impact upon programs:

- Reliability: The probability that a system, subsystem, component, or device will satisfactorily perform its intended function without failure, for a prescribed period of time, within a prescribed environment.
- Confidence Level: The probability that the reliability figure of merit predicted for a system, subsystem, component, or device is correct.

5.1.1.3 The Statistical Implications. Traditionally, the reliability of any system is determined by mathematical techniques prior to the ultimate deployment of the system for use; and the confidence level associated with that figure of merit arises from statistical treatment of the test results obtained in the course of verifying the system function and life potential.

Expressed in non-statistical terms, a typical reliability specification for a space vehicle and an associated confidence level may be cited as follows: The space vehicle reliability shall be 85% and this shall be demonstrated to a 90% confidence level; the on-orbit operational period to be one year. This means simply that if 100 absolutely identical spacecraft were built, no more than 15 of them should fail for any cause while on-orbit; and, based on data derived from tests prior to the flight of the first spacecraft; one may be 90% sure that the 15 failures expected will not be exceeded during the one year orbital period.

As a matter of statistical purity, it would take 17 spacecraft performing without any failure during a one-year period to demonstrate an 85% reliability to a 90% confidence level; therefore, the reasons for using statistical manipulation of pre-flight test data to demonstrate compliance are obvious, in view of the very large cost impact of a true demonstration based upon the ratio of failures to successes in a given number of trials.

5.1.1.4 Spacecraft Reliability/CL in the Shuttle Era. Because historically high confidence in the high reliability of spacecraft has been considered essential, program managers have required values for both reliability and confidence level that are the highest attainable within the overall program budget. In the late 1970's with the Shuttle available, and in-orbit maintenance possible, reliability values and associated confidence levels can be lower. Practically, it does not seem desirable to assign reliability values and confidence levels as low as R = 0.50 C.L. = 50% since such assignments would permit equal chance for failure and success, and predictions would have as much chance of being incorrect as correct (although programs designed to such criteria would be relatively inexpensive). It does appear attractive, however, to assign reliabilities of a slightly higher order, and confidence levels which bias the odds more toward success. Accordingly, for low-earth-orbit type spacecraft, reliabilities of R = 0.600 and confidence levels of C.L. = 60% afford the lowest cost approach to potentially effective systems.

Communication and other satellites in Syneq orbits are somewhat more difficult to emplace and revisit and require the services of a Space Tug for movement between LEO and Syneq orbit. Because orbit lifetimes tend to be longer and onboard subsystems tend to be more sophisticated for these Syneq orbit spacecraft, their reliabilities and confidence levels must be further biased toward success. After considerable examination, the most cost effective values appear to be R = 0.75 C.L. = 70%.

Cluster spacecraft will require regular visits for maintenance and experiment updating. A visit schedule of twice per annum during the orbital period seems feasible, and during such visits maintenance to offset on-orbit failures can be undertaken, as well as routine experiment updating. For Cluster spacecraft, the reliability and confidence levels applicable to the other LEO types of spacecraft appear satisfactory.

Planetary spacecraft benefit from the Shuttle in that they can be checked out and necessary repairs made after the launch/ascent phase and prior to final dispatch from earth orbit to the planetary destination. Thereafter, no further attendance is possible, and as the on-board subsystems are usually sophisticated, it appears feasible to assign reliability and confidence values identical to those assigned to Syneq spacecraft. Figure 5-1 summarizes the reliability values and confidence levels applicable to various spacecraft/payload types.

Hardware	Missions	Reliability			C.L. (Lower Bound.)	MMD	1st Probable Failure
		P/L	Exp.	S/C			
Mission-Peculiar Spacecraft	LEO	.60	.92	.65	60%	2 yr.	11-13 mo.
	Syneq	.75	.95	.79	70%	5 yr.	28-32 mo.
	Plan.	.75	.93	.81	70%	2 yr.	11-13 mo.
Cluster Spacecraft	LEO	-	-	.65	60%	1 yr.	4-6 mo.

a. MMD = Mean Mission Duration = The expected hardware operating life during which the system will function satisfactorily given reasonable maintenance. A full refurbishment will be assumed following the MMD period.

b. 1st probable failure = The point in mission time that the 1st major failure will probably occur.

c. "P/L" = payload; "Exp". = Experiment; "S/C" = Spacecraft.

Fig. 5-1 Characteristics of Shuttle-Era Spacecraft

Summarizing, the Shuttle permits on-orbit repair and maintenance of spacecraft. Maintenance on-orbit permits higher risks of failure to be taken, and the failures can be offset by using the Shuttle as a service vehicle to facilitate initial repairs, remedial maintenance, and full refurbishment of spacecraft. If higher risk of on-orbit failure is permissible, the reliability and confidence values can be lowered, and spacecraft can be designed for maintenance; then the reliability values will apply for the on-orbit periods between maintenance visits during which time the spacecraft will be unattended. Such maintenance intervals will be shorter than the full service life of an expendable spacecraft. Such techniques permit a maintenance schedule to be set up, and an allowable failure budget for each maintenance interval to be established.

5.1.2 Cost-Affecting Tradeoffs

5.1.2.1 Effects of Reducing Reliability and Confidence Level. Reducing reliability requirements and the associated confidence levels drive costs down in the following areas:

- RDT&E: Designs can be simpler due to the fact that less redundancy will be required to offset potential failures. Designs can be less costly due to the fact that equipment must work only for the period between maintenance visits rather than for the full spacecraft operational life.

- Testing: Development and qualification testing can be reduced in duration and complexity due to the fact that tests conducted to verify failure-free function and life capability are directly affected by the reliability value and the confidence level to which the reliability must be demonstrated. The lower the reliability and confidence level, the fewer the number of tests required, and the shorter the duration of the test phase.

- Manufacturing: Due to the large payload weight and volume capability of the Space Shuttle, packaging of hardware into modules can be effected using lower density packaging, greater weight, and greater volumetric dimensions. Low cost design techniques discussed in Section 7 thus apply.

- Pre-Launch Checkout: Can be simplified, due to the simpler design of the hardware.

- Immediate Post-Launch/Ascent Phase: As a direct result of using the Shuttle, payloads can be checked out, and repaired if required, after ascent, prior to final deployment. Subsequent to deployment, the Shuttle loiter capability permits payload retrieval and repair as necessary in cases of early-life malfunction.

5.1.2.2 Tradeoff of Transportation Costs, Spacecraft Costs, and Reliability. Against the positive cost-saving advantages of low-cost, lower-reliability payloads must be compared the fact that maintenance on-orbit requires a Shuttle flight in the case of an LEO type payload, and a Shuttle and Tug flight for a Syneq type payload. The costs of transportation for a Shuttle flight have been estimated at $7.3 millions, and those of a Shuttle-Tug combination at $7.9 millions; to which must be added the costs of spares modules carried to orbit to effect the repair action.

For example, it is not economically attractive to maintain a low-cost spacecraft with a unit cost of $5.0 million or less, which requires three repair visits during the orbital period at $7.3 millions per visit, plus the costs of spare modules. However, the on-orbit maintenance of a payload costing $20.0 millions per unit, and requiring only one maintenance visit per orbital period is very attractive economically. The cost of the spacecraft and its periodicity of maintenance (as a function of its reliability) are factors which must be traded off before assigning reliability and confidence levels to a payload program, and specifying the degree to which test demonstration of these characteristics must be made.

5.2 Cost Impact of Payload MMD

5.2.1 Definitions for MMD

Mean Mission Duration (MMD) may be defined as: that expected, or mean mission time that a system will perform satisfactorily without resupply or maintenance, considering all factors. MMD may be defined mathematically as the area under the reliability-time curve from time zero to mission truncation time or to the end of the program, whichever is first. The term and its definitions apply specifically to historical expendable spacecraft where no maintenance has heretofore been possible. Practically, while the term does not apply to spacecraft designed for on-orbit maintenance or on-orbit retrieval and transport to earth for ground based maintenance, it has been used for the sake of convenience. In this revised context, MMD denotes that time period of on-orbit operation at the end of which the spacecraft requires full refurbishment to continue normal operation. Thus, MMD may be construed to mean the service or useful operating life of the spacecraft. MMD is expressed in years, and is the life parameter which the spacecraft must satisfy. As an example, a spacecraft having a 2 year MMD must operate satisfactorily for that period on orbit. Should failure occur prior to the MMD point, repair action, which is confined to removal and replacement on-orbit of the equipment module(s) exhibiting failure, must restore normal operation until the next failure, or the MMD point, whichever is first. At the MMD point, all functional modules of the spacecraft will be replaced, the only elements retained being the spaceframe and the integral wiring harnesses.

5.2.2 Effect of MMD on Payload Program

MMD selection has a considerable impact on costs in both the RDT&E and flight operations phases of a space program, and interacts with the reliability requirement directly. In general, the longer the MMD, the greater the mission costs, as evidenced by the following:

a. MMD vs Spacecraft Hardware

If the reliability value is held constant, doubling the MMD value halves the failure budget allowable to achieve a given reliability. If the allowable failure budget is halved, then more hardware redundancy must be added in order to assure failure free operation of the payload on orbit.

b. Test Time vs MMD

While the RDT&E testing to demonstrate a given reliability at a given confidence level does not change with respect to the MMD in terms of the required success-to-failure ratio, the duration of each test changes considerably. For example, if it is presumed that a test of 10% of the desired on-orbit time is conducted on earth, under simulated space conditions, to provide data from which inference of life capability can be made; and if the MMD requirement is doubled, then the test duration time must be doubled.

5.2.3 Tradeoffs of MMD and Repair/Refurbishment

If the costs of requiring a spacecraft to have an MMD of 5 years and fulfill a mission of 5 years duration are justifiable, then no refurbishment need be undertaken at the mission end point. However, there is a finite probability that a repair visit will be required at, or about the mid-MMD point. Thus, the number of Shuttle flights would be two, one to initially place the payload and another to repair the failure (if it occurs). If the mid-MMD chance of failure applies equally to a spacecraft of one-year MMD assigned to perform a 5-year mission, then the number of flights would be one placement, 5 repair flights, and 4 refurbishment flights, for a total of 10. The costs of the one-year MMD payload may be considerably less than that of the 5-year payload in terms of RDT&E costs and unit costs, but this advantage may well be offset by the overall transportation costs.

Examples of such tradeoffs are shown in Figs. 5-2a and 5-2b for LEO-type missions, and in Figs. 5-3a and 5-3b for Syneq-type missions. The balloons and rectangles on Figs. 5-2b and 5-3b indicate the lowest-cost points. Such tradeoffs should be undertaken to determine the optimum MMD selection for any given mission profile.

The cost summaries in Figs. 5-2b and 5-3b show basic comparisons between a Shuttle-launched expendable payload and a refurbishable payload. The modes are described following:

		Spacecraft MMD		
		1 Yr	2 Yr	3 Yr
Spacecraft RDT&E Cost	Expendable Refurbishable	$ 95M 95	$ 105M 107	$ 115M 120
Spacecraft Unit Cost	Expendable Refurbishable	15 17	15 17	20 23
Transportation Cost	Initial Placement Spacecraft Replacement Repair or Refurb.		3.7 M(1) 3.7 M(1) 2.4 M(2)	
Spacecraft Spares (Average) Repair / Refurb	- -	2.2 M per set 4.0 M per set		

(1) Assumes Shuttle flight shared between 2 missions = 50% of $7.3 Million
(2) Assumes Shuttle flight shared among 3 missions = 30% of $7.3 Million.

Fig. 5-2a MMD vs Cost Tradeoff - LEO Mission

S/C MMD (Yr.)	Cost Category	Payload Program Cost ($ Million)							
		Expendable S/C (1)				Refurbishable S/C (1)			
		Mission Duration (Yrs)				Mission Duration (Yrs)			
		1	2	3	10	1	2	3	10
1	RDT&E	95	95	95	95	95	95	95	95
	Unit S/C	30	60	90	285	17	17	17	17
	Spares	-	-	-	-	2	9	15	56
	Transportation	7	15	22	74	6	11	16	50
	Total	132	170	207	454	120	133	143	218
2	RDT&E		105	105	105		107	107	107
	Unit S/C		30	45	150		17	17	17
	Spares		-	-	-		5	9	27
	Transportation		7	11	37		6	9	25
	Total		142	161	292		235	242	166
3	RDT&E			115	115			120	120
	Unit S/C			60	140			23	23
	Spares			-	-			2	19
	Transportation			7	26			6	18
	Total			182	281			151	180

(1) Shuttle-launched

Fig. 5-2b MMD vs Cost Tradeoff - LEO Mission

		Spacecraft MMD						
		1 Yr	2 Yr	3 Yr	4 Yr	5 Yr	7 Yr	10 Yr
Spacecraft RDT&E Cost	Expendable Refurbishable	50 55	55 60	65 75	70 80	80 90	125 150	175 200
Spacecraft Unit Cost	Expendable Refurbishable	5.4 5.5	6.0 6.5	6.8 8.0	8.5 9.0	9.0 9.5	14.0 15.0	19.5 20.0
Transportation Cost	First Placement S/C Replacement Repair or Refurb	$ 7.9M for Shuttle + Tug (7.3 + 0.6) $ 7.9M for Shuttle + Tug 1 Flight Shuttle + Tug can service 4 Spacecraft						
Spacecraft Spares Cost (Average) — Repair Refurb	- -	$ 2.2 Million per set (incl. residuals) $ 4.0 Million per set (incl. residuals)						

Fig. 5-3a MMD vs Cost Tradeoff - 10 Yr Syneq Mission

	Mission Duration (Yrs) and Cost ($ Million)						Comments
MMD	1	2	3	4	5	10	
1	M $\boxed{\$126}$ E 156	M $175 E 277	M $183 E 374	M $185 E 475	M $207 E 582	M $519 E 992	
2		M $\boxed{150}$ E 166	M $\boxed{155}$ E 222	M 186 E 278	M 199 E 333	M 336 E 556	
3			M 163 E 182	M 169 E 242	M 196 E 301	M 283 E 406	
4				M $\boxed{164}$ E 206	M 196 E 272	M $\boxed{242}$ E 278	
5					M $\boxed{174}$ E 215	M $\boxed{242}$ E 278	
10						M 323 E 318	M = Maintainable S/C (2) = $\boxed{XX}$ E = Expendable S/C (3) = XX

(1) All payloads Shuttle-launched. Costs include Shuttle/Tug transportation costs, but exclude payloads operations costs.

(2) Repair and partial refurbishment accomplished at intervals of MMD/2; full refurbishment accomplished at intervals equal to MMD.

(3) New spacecraft launched at intervals of MMD/2 (at first major failure occurrence).

Fig. 5-3b MMD vs Cost Tradeoff - 10 Yr Syneq Mission

a. Expendable Payload

The initial payload is launched and operates satisfactorily until the first major failure occurs, at approximately MMD/2. A replacement payload is launched to replace the first. The first is neither retrieved nor refurbished. This cycle is repeated at intervals of MMD/2 for the total mission duration specified.

b. Refurbishable or Maintainable Spacecraft

The initial payload is launched and operates satisfactorily until the first major failure occurs at approximately MMD/2. A Shuttle flight to orbit carries sufficient spacecraft modules to replace the failed module and those that have been predicted to fail in the near future (partial refurbishment or preventive maintenance). A full refurbishment will be accomplished at the MMD point (full refurbishment can be undertaken early should definite indications of accelerating rate of failure be received from the orbiting payload; also, repairs can be made or not, dependent upon the nature and severity of the actual failures occurring).

In all the cases surveyed for LEO missions the refurbishable-spacecraft mode was less costly than the expendable-spacecraft, for MMD's of 1, 2 or 3 years and for mission durations of 1 through 10 years. The lowest program cost occurred with a spacecraft MMD equal to one year.

In analyses of Syneq spacecraft, spacecraft MMD's of 1 through 5, and 10 years were surveyed for mission durations of 1 through 10 years. For a 5-year mission, a refurbishable spacecraft with 5-year MMD offers the lowest-cost program. For a 10-year mission, a refurbishable spacecraft with a 4 or 5 year MMD provides minimum cost.

5.2.4 Failure Occurrence vs Orbit Maintenance

5.2.4.1 Historical Emphasis on Minimizing Failure Probability. Traditional reliability techniques applicable to space payloads have presumed that a spacecraft of R reliability and a failure potential of Q, (where $Q = 1 - R$) may, or may not exhibit failure during operation. Should failure occur it will be random in nature. Within accepted reliability practice, the methodology used has been to make the probability of failure Q as small as possible with respect to the reliability R; a practice which has been extremely costly in terms of hardware redundancy incorporated to offset any, and all failures, and in terms of tests to verify that the required high reliability has been designed into the system.

5.2.4.2 Analysis of Historical Spacecraft Failure Data. During the course of a large number of US space programs, considerable data concerning the mechanisms and characteristics of failure have been amassed. Figure 5-4 illustrates a composite of such data.

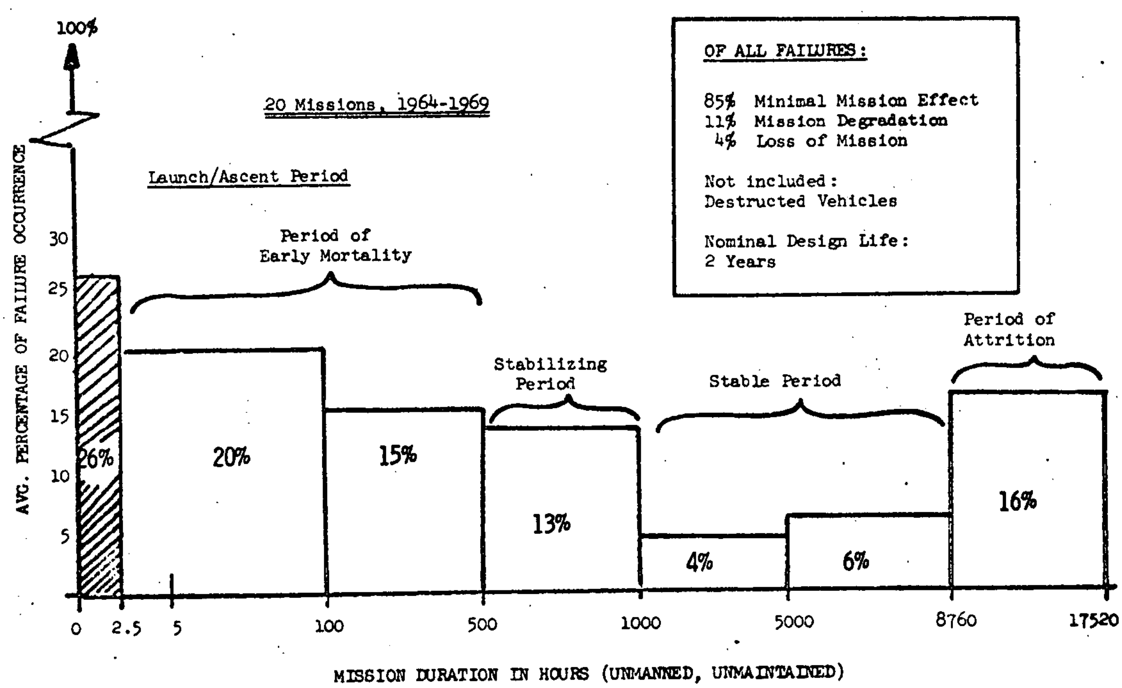


Fig. 5-4 Percentage of Failures vs Mission Duration

5.2.4.3 Potential Elimination of Launch/Ascent and Early-Life Failures by Shuttle. It may be noted that on missions of 2 years duration, 84% of all failures reported occurred prior to the mid-MMD point of 8,760 hours. With the Space Shuttle able to provide pre-placement payload checkout, and also immediate post-placement recovery and repair capability, it appears reasonable to expect that the 46% early life failure incidence might be eliminated.

5.2.4.4 Orbit Maintenance to Accommodate Other Failures. The remaining 38% of the failures is still an appreciable number of failures occurring prior to mid MMD, and with new generation spacecraft of the maintainable type, having relatively low reliability, there is a very real probability that if an orbit failure occurs it will occur near the mid-MMD point. By means of such statistical techniques as the Weibull Mortality Equations, and for electrical/electronic hardware, the Poisson Approximation Equations; it is possible to compute the probability of exactly one, one or more, or one or less failures at the mid-MMD point. Where the probability is appreciable, as it will be in cases where the spacecraft overall failure-potential Q is of the order of 40% for LEO spacecraft or 25% for Syneq spacecraft; a decision can be made to schedule a repair flight at that point in time. Even if the failure has not exhibited itself at the anticipated time, the flight can be used to replace those subsystem modules which theoretically are approaching a failure point computed by the means stated. The rationale is thus set up for (1) repair where a failure has occurred, and (2) on the same repair flight, partial refurbishment by replacement of those modules which have not failed, but for which failure is imminent. Full refurbishment at the MMD point, wherein all the modules are changed, gives the spacecraft a new lease on life. (The statistical methods outlined are not detailed herein for the sake of brevity, but are well known and proven techniques and can be found in the majority of statistical references).

5.3 Benefits of Hardware Standardization to Payload Repair/Refurbishment/Reuse

There are several advantages to hardware standardization which give rise to appreciable cost benefits to space programs. These cost benefits apply throughout all phases of programs and are not limited to the on-orbit phase. Standard hardware may be of three forms (described in detail in Section 4):

a. Standard Modules for a "family" of spacecraft with similar missions

b. Standard Spacecraft designed to perform a number of similar missions

c. Cluster Spacecraft designed to support several mission experiments simultaneously.

5.3.1 Cost Advantages Arising from Standard Modules for On-Orbit Maintenance

Where more than one spacecraft are to be placed by a single Shuttle flight, a case envisioned as likely for the LEO Earth Observatory Satellite type programs, savings can be realized by such transportation sharing. Consider a Shuttle flight which is to emplace two EOS type spacecraft. In this case, a classic approach would require that not only two spacecraft must be carried, but also a complete set of spare modules for each, to offset the potential incidence of Infant Mortality failures. Historically, such incidence has been high as previously discussed in par. 5.2.4. The spare module complement, however, need not be two full sets, one set per spacecraft.

Based upon the usual Product Rule for systems, which gives very conservative results, and the technique of Complexity Ratio for apportionment, a typical EOS payload might have a reliability budget as follows:

R Payload			0.60
R Experiments		0.922	
R Spacecraft		0.652	
Subsystems	R Structure	0.999 +	
	R ECS	0.999 +	
	R CDPI	0.864	
	R ACS	0.997	
	R S&C	0.850	
	R Electrical Power	0.889	

With the subsystems allocations further apportioned among the several modules per subsystem, the reliability numbers become higher still. Selecting a representative module such as the sensor module from the S&C subsystem, the reliability is in excess of 99% due to the hardware and functional redundancy included. The failure probability of one or less percent is thus not great, and applies to each of the two modules within the S&C subsystem of each of the two spacecraft carried by the Shuttle.

Should this module fail in Spacecraft No. 1 either prior to the pre-placement checkout onboard the Shuttle or immediately post-placement within the Shuttle loiter period, the laws of chance have resulted in failure within the 1% Q domain. With respect to the identical module in spacecraft No. 2, the chance of not failing is still 99% and, while the failure is possible as a second incidence of the same anomaly, the likelihood is not greater than 1% and is probably less.

It would appear reasonable, therefore, not to carry more modules than one of each type, i.e., one set to accommodate both spacecraft. If greater assurance is required, then the technique to determine the optimum number would be to rank all modules in descending order of their probability of failure and carry two each of the modules having the greatest failure potential. In this case, there would be very few modules required at a level of more than one spare module for each module type.

5.3.2 Refurbishment/Reuse of Modules

Assuming that the maintenance-on-orbit philosophy is adopted, modules which have failed, modules approaching failure, and modules which have operated to their theoretical MMD point will be returned to earth for major refurbishment. Detailed examination of these modules suggests that refurbishment at all hardware levels down to the piece part is cost effective, and that the spacecraft subsystem modules have a high residual value. While the residual value varies from module to module with a high of 90% and a low of 45%, the average residual value appears to be about 75%. Thus the cost of refurbishing most modules is approximately 25% of the initial purchase price. It is obvious that the cost of modules procurement can be greatly reduced by taking advantage of the comparatively low refurbishment costs and the high residual value of the modules. Some examples of ground maintenance concepts are:

5.3.2.1 Examples of Refurbishment. Some examples of ground refurbishment approaches are represented by the following:

a. Module Refurbishment

The first echelon of ground refurbishment is at the module level. Figure 5-5 is an abbreviated flow diagram for a typical Communications module.

b. Sub-Assembly Refurbishment

Following removal of subassemblies from the used/failed module, each sub-assembly is processed through a refurbishment cycle. Figure 5-6 is a flow diagram for a typical Stabilization & Control Electronics package.

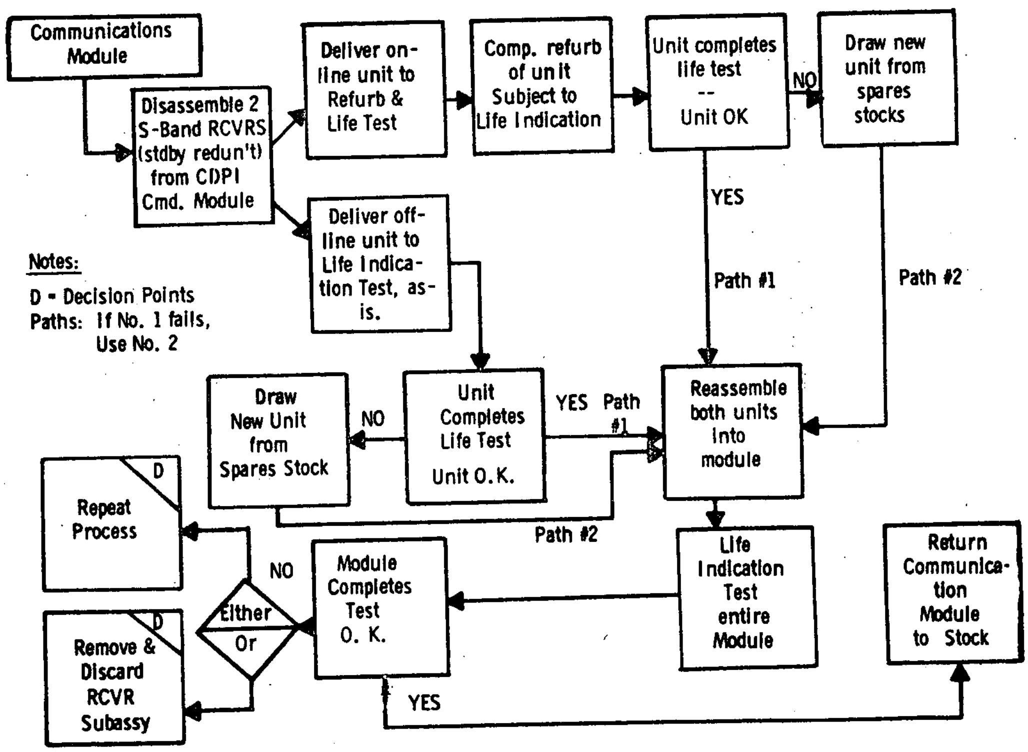


Fig. 5-5 Typical Module Refurbishment Flow Diagram - Communications Module

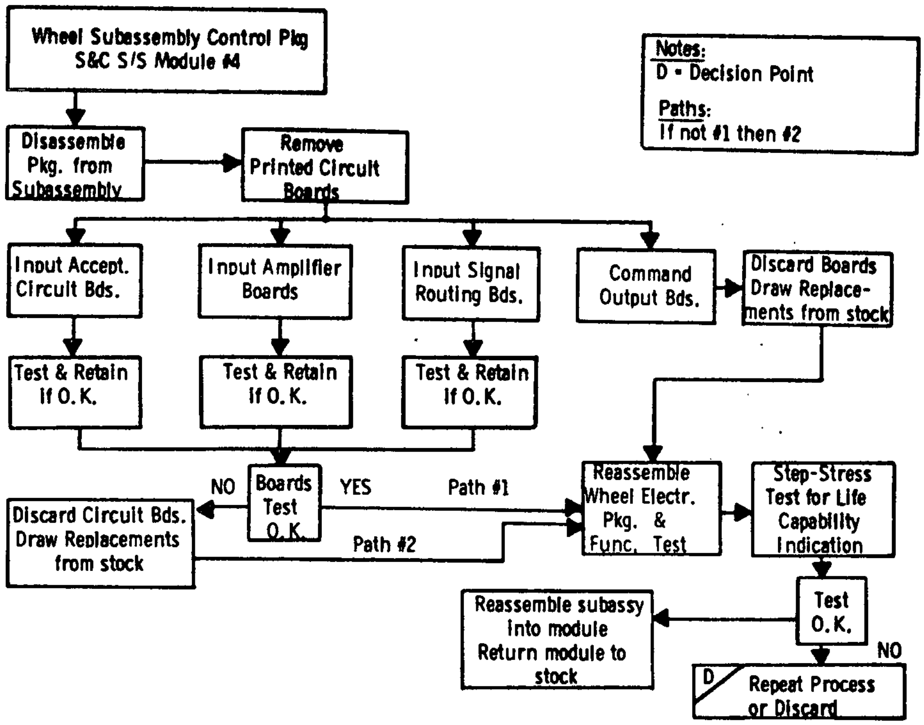


Fig. 5-6 Typical Subassembly Refurbishment Flow Diagram - Wheel Control Electronics

c. Component Refurbishment

Most of the components removed from the modules or subassemblies are also refurbishable and have significant residual value. A flow diagram for refurbishment of a typical electromechanical component is shown in Fig. 5-7.

Sets of new replacement parts for component refurbishment would be procured initially and held in bonded stores at field refurbishment depots awaiting the start of the refurbishment cycle on a particular program.

5.3.2.2 Cost-Savings with Ground Refurbishment. Two examples of the cost savings available with use of ground refurbishment and reuse of spacecraft hardware are provided below.

a. Valve Replacement vs New Unit Replacement

A comparison of the costs of refurbishing a solenoid valve versus replacing the used/failed valve with a new unit is shown on Fig. 5-8. The total cost to refurbish the valve is $314, including amortization of special test equipment; the unit cost of a replacement is $1958. Applying a conservative factor, the actual cost of the refurbished valve is only 26% of the new cost.

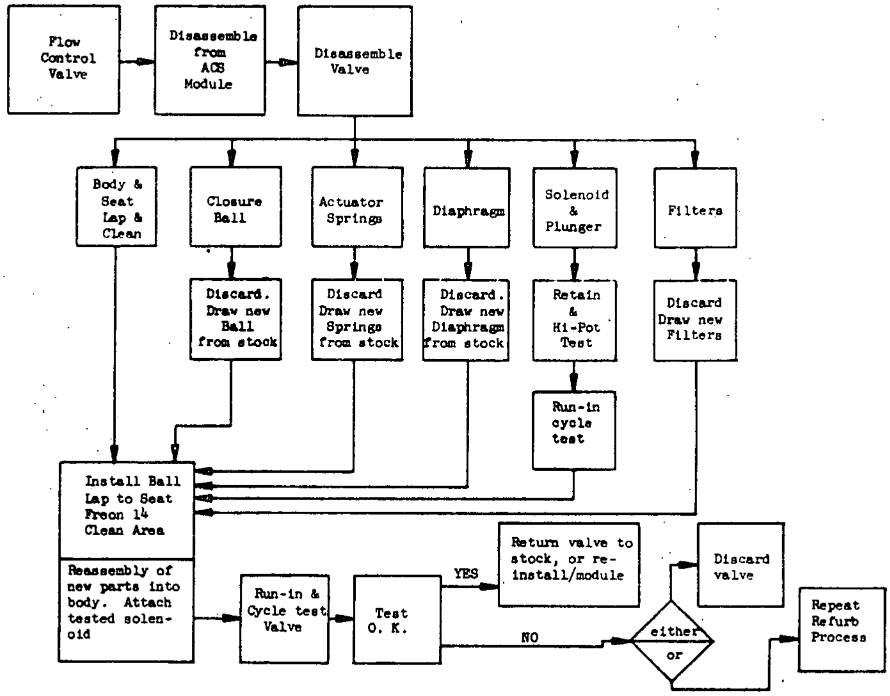


Fig. 5-7 Typical Component Refurbishment Flow Diagram - Flow Control Valve

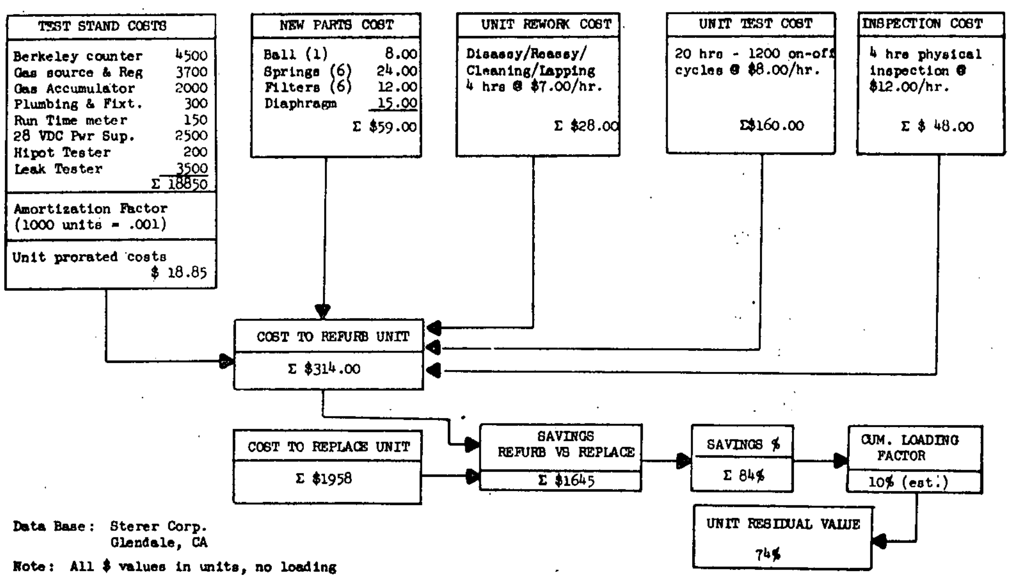


Fig. 5-8 Cost of Solenoid Valve Refurbishment vs New Unit Replacement

b. Typical Module Refurbishment Costs

The cost of complete module refurbishment has been derived, the combined effects of part replacement and component and subassembly refurbishment. Three typical spacecraft modules are listed on Fig. 5-9 with the costs of refurbishment.

Sub-System Module	Hdwr Refurb Level	Maintenance Action	Replacement Elements	Cost		Repl Hdwr Type	Cost of Refurbishment ($)					Savings *		
				Repl. Elem.	Initial		Repl Parts	Dissy Reasy	Test	QA Insp.	Total	$	%	Res. Value %
ACB 1-4	Component	Remove valve from module. Refurb & test	Valve, flow Reg.	$1958	$1958	Parts Ball Spring Filter Diaphr. Sol'nd	 8. 24. 12. 15. retain 59.00	28.	179.	48.	314.	1645	16.	84.
S&C Wheel Mod 4	Sub-assy	Remove wheels & gimbals. Remove elec. control pkg. Disassemble wheel units & refurb & test	Wheels pkg. compts: Wheels Motors Electr pkg.	 $300K 18K 77K	$395K	Brgs Gimbls Bushgs *Mtrs *Mltr	6K 18K 2K 15K 3K 44K	280.	4000.	500.	48780	345K	12.4	87.6
CDPI Comm	Module	Remove both RCVR units from modl. Off-line unit direct to test On-line unit refurb by replacing C/B's then life test	S-Band RCVR's (2)	$12.5K per unit	$25K	Pwr & control C/Bs 2 per RCVR (4 total)	only unit 2 $800	only unit 2 $21.	Unit #1 $2400. Unit #2 $2400.	360. 360.	$6341 both units total	18.7 K	25.5	74.5

* Costs to refurbish & reinstall motors & electronic control package.
Note: All $ values are unloaded direct costs in units.

Fig. 5-9 $ Savings for Module Refurbishment vs Replacement

5.3.3 The Impact of Hardware Standardization Upon Module Refurbishment/Reuse

The foregoing plan for module refurbishment/reuse and the large payoff in dollar savings (versus replacement of used/failed spacecraft modules with new modules) can be enhanced even further with standardization of the hardware elements; parts, components, and the modules themselves.

Not only will the procurement and warehousing of spare parts be simplified, but the variety of field crew used for disassembly, assembly, repair, and retest can be substantially reduced.

Further, service experience with standardized hardware can be more readily accumulated, diagnosed, and corrective action initiated. A much larger universe for test sampling of identical or similar hardware will be available and both ground test and flight article experience can be combined into a centralized set of operational data which will:

a. improve the confidence level in the hardware (allowing early reduction in scope and depth of ground testing)

b. allow lowering the reliability goals (with attendant cost reduction) for follow-on or new similar spacecraft.

5.4 Effective Combinations of MMD/Repair/Refurbishment

As may be seen from the foregoing text; reliability, MMD, Mission Duration, on-orbit repair, partial refurbishment, full refurbishment, and selective ground maintenance at several levels to refurbish modules returned from space all have strong and interacting impacts upon costs to a payload program. In addition, the level of confidence to which reliable performance is to be demonstrated has a decided impact upon the cost of performance-verification-testing; further, in cases where overall mission success is to be demonstrated at the end of a mission (or missions), confidence level has a cost impact on the planning and testing of hardware destined for the next similar mission. As all of the parameters mentioned are interactive and variations can be made to achieve cost effective compromises, iteration of such variations will indicate several trends which permit general conclusions to be made. These general conclusions are of value to the planning of space programs using either mission-peculiar or standard hardware or both. The main conclusions may be listed as follows:

- In selecting reliability figures of merit and confidence levels select minimum system values consistent with mission requirements.

- Tradeoff MMD versus mission duration, with selected reliability held constant, so that the optimum program cost, including transportation, repair, refurbishment, and modules residual value can be tabulated and compared (as shown in Figs. 5-2b and 5-3b.

- Design modules for subsystems which exhibit the greatest residual values (perform analysis as illustrated in Figs. 5-8 and 5-9).

- Investigate the possibility of transportation sharing for both placement and maintenance flights.

- Pre-schedule Shuttle flight operations to perform both repair and full refurbishment of orbiting payloads (nominally, repair at MMD/2; refurbishment at MMD point).

- To offset launch/ascent and early-life failures, carry a set of spare modules to orbit with each initial-placement spacecraft. The particular types and quantities of modules will be based on statistical analyses of failure probability.

- For Cluster spacecraft schedule revisits at half yearly intervals; and at the time of experiments change, recalibration, or updating make whatever spacecraft repairs are indicated and partially refurbish in accordance with mid-MMD potential failure expectation.

- When undertaking the refurbishment of modules returned from space, set service life limits upon the modules; i.e., after n refurbishment operations during which some of the module internal hardware elements are retained, the module should be considered as expendable. The discard point should coincide with the service life of the longest-lived element (dynamic) of the module under consideration.

The techniques set forth in this section as being productive of cost effective space hardware and programs have not been detailed to any extent. The concepts appear viable after subjecting them to limited application. Further study is required to refine these concepts into a complement of estimating methods by which informed cost decisions can be made.

Section 6

APPLICATION OF LOW-COST APPROACHES TO PAYLOAD PROGRAMS

6.1 General Application of Low-Cost Payload Program Approaches

In the development of the concept of spacecraft hardware standardization, it has been very apparent that application of low-cost design approaches is a natural corollary. Many of the spacecraft and program approaches demonstrated as cost-effective during the earlier Payload Effects studies are equally applicable when considering standardization. The principal features of the low-cost methodology are provided in the following paragraphs.

6.1.1 Mandatory System Performance and Design Requirements

Over-specification has been a significant factor in escalating the costs of historical space programs, beginning with the System Performance and Design Requirements Specification prepared by the government program office and/or the contractor. Cost consciousness must be fostered among the scientists, engineers and managers responsible for program planning, with the objective of obtaining requirements specifications based on cost/value analysis.

6.1.2 Mission Requirements

Frequently, the basic mission requirements are somewhat arbitrarily established without consideration of the impact on the spacecraft or program costs. Many "desired" objectives are initially included in the program and soon become mandatory. Tradeoffs of the mission objectives against cost to implement should always be employed; reduction, if not elimination, of certain non-mandatory requirements can have a significant program cost impact.

6.1.3 Simplified Equipment Specifications

Engineering organizations currently prepare design specifications that are more restrictive than they need to be, because it is safer (but more costly) to err in that direction. Program offices must encourage engineers to specify the lowest acceptable equipment performance and other design requirements, and effective review procedures must be established to control over-specification of requirements for both in-house fabricated and procured equipment. Preliminary analyses should be required to backup any super-restrictive requirement.

6.1.4 Simplified Engineering Documentation

Much of the escalation of the costs of space programs is attributable to the documentation requirements. Program planners should carefully evaluate documentation needs, and impose only those requirements essential to the orderly execution of the program. In particular, they should avoid the imposition of requirements simply because it is safer to require too much and avoid the risk of requiring too little.

When contractual documentation requirements permit, simplification of contractor engineering documentation can result in significant savings.

6.1.5 Special Reliability Considerations

6.1.5.1 Cost/Reliability Relationship. It is well known that the cost of raising the predicted reliability of space vehicle systems and subsystems above nominal values increases exponentially as the ultimate reliability of 100 percent is approached. Part selection becomes more rigorous, reliability testing becomes more comprehensive, and more and more redundant components and backup functions are required. For example, a typical cost estimating relationship (CER) showing the relationships between cost and reliability for a typical space payload subsystem is presented in Fig. 6-1.

The exponential increase of cost as reliability is increased beyond 0.75 (75%) is most significant. If the reliability required of a collection of complex subsystems (spacecraft) can be limited to about 0.7, appreciable savings can be realized in the design, development, and production of space payloads. Such limitation of reliability requirements will be possible for Shuttle-launched payloads because of the capability of the Shuttle to checkout payloads in orbit prior to deployment, to repair them in orbit, and to recover them for reuse.

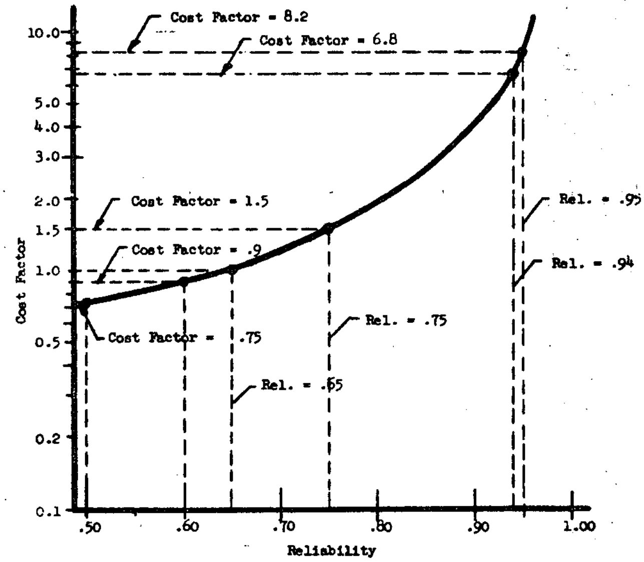


Fig. 6-1 Cost/Reliability Relationship for Typical Historical LEO Payload

6.1.5.2 Predicted vs Achieved Reliability. Justification for lower system reliability requirements for Shuttle-launched payloads is afforded by analysis of the historical disparity between the predicted reliability of expendable-launched payloads and the reliability they have actually achieved in orbit. Examples are given in Fig. 6-2.

Program	Predicted Reliability	Achieved Reliability
LMSC Small Research Satellite (SRS)	< 0.80	0.90
HEOS-A, German	0.85	0.98
Lunar Orbiter	0.75	0.90
Mariner	0.75	0.90

Fig. 6-2 Predicted Reliability vs Achieved Reliability

The disparity is partially explained by the following:

a. There has historically been no differentiation between major or critical failures and minor failures in the failure-rate "bibles" such as FARADA. Inclusion of the minor failures (which do not cause payload nor mission failures) in the total failure statistics has biased the resulting spacecraft designs to conservatism and higher costs.

b. There has historically been no method by which launch/ascent and early-life failures could be eliminated from the total payload mission failure. A highly conservative factor was therefore assigned by the spacecraft designer for the "unknown" launch/ascent phase of operation. Typical historical conservatisms are shown on Fig. 6-3. (These are factors applied to laboratory ambient.)

Environmental Exposure	Adversity Factor
Launch Operations	$\boxed{90}$ (1) (2)
Aircraft Operations	$\boxed{18}$
Road Transport	15
Rail Transport	5
Shipboard Operations	3
Submarine Operations	2
Space Operations	$\boxed{1.5}$

(1) Some Contractors/Agencies believe launch stress should be 900-1500; others use as low as 4. LMSC uses 10 max. for solid-state hardware, 35 max. for others.

(2) This factor is not time dependent.

Fig. 6-3 Adversity Factors for Payload Reliability Prediction (Data from MIL-HDBK-217A)

The steady-state space environment, once attained, is rated a factor of 1.5; the launch/ascent phase is rated a factor of 90, with various agencies or contractors using factors from 10 to over 1000.

6.1.5.3 Methods for Lowering Reliability Requirements. Section 5 provides discussion on (1) the latest failure statistics, segregating the launch/ascent phase, and (2) the use of the Shuttle in performing on-orbit checkout and repair of the payload following the launch/ascent. The application of reasonable (major or critical only) failure statistics, the segregation of on-orbit and launch/ascent failures, and the use of the Shuttle pre-deployment repair will all contribute significantly to reducing the payload reliability requirements. These new principles should be applied in reducing design, development, and testing costs.

6.1.6 Reduced Testing Requirements

Typically, spacecraft equipment and assembled spacecraft have been subjected to very comprehensive, rigorous, and often repetitive testing programs to establish the level of confidence felt to be necessary for the making of launch decisions. Such programs have been costly to execute and often costly in worn-out or damaged equipment. The fear of spacecraft failure during launch or early in orbital flight and of the consequences of such a catastrophe has been the principal motivator for traditional testing programs. Much of this fear will be alleviated by the Space Shuttle which makes possible in-orbit checkout and repair of spacecraft before they are committed to orbit, monitoring of their early orbital performance, and their recovery for repair in orbit or their return to earth in the event of early malfunction. Once fears of the consequences of failure have been alleviated it should be possible to reduce significantly the scope and cost of testing programs.

6.2 Considerations and Guidelines for Payload Low-Cost Design

Historically, the design of spacecraft has been severely constrained by weight and volume limitations resulting from the modest payload capability of the launch vehicles. When weight and volume constraints are relaxed, significant cost reductions in spacecraft can be made. The following paragraphs offer some approaches.

6.2.1 Design Simplification

Engineering costs are directly related to the complexity of the designs to be created. To reduce engineering costs, payload designers must strive for design simplification, beginning in the concept design phase and continuing through the detail design phase. Simple designs require fewer and simpler specifications and drawings, less analysis, less manufacturing and assembly labor, less testing and less integration and coordination. Select a simple spacecraft configuration which requires only a simple structure.

6.2.2 Low-Cost Materials and Fabrication Methods

In general, for comparable applications, materials that are inexpensive to buy and fabricate, such as steel and in certain cases, aluminum, are usually heavier for the same application than titanium and beryllium, which are expensive. In the past, payload weight limitations have prevented the free use of the cheaper materials and simple methods of fabrication; and the costs of spacecraft have been higher than they would have been without weight limitations. In designing spacecraft that will be orbited and supported by the Space Shuttle, full advantage should be taken of the great payload capability of the Shuttle, and low-cost materials and fabrication methods should be employed almost exclusively. Use commercially available grades/sizes of aluminum sheet and extrusions for most structural elements. Do not use beryllium, composites, or other high-cost materials.

6.2.3 Use of Pre-Qualified Equipment

Most space programs have made some attempt to use pre-qualified equipment in new designs to obtain the obvious cost savings. However, existing qualified equipment has often been considered and rejected because it was not optimized functionally, or in size and weight, for the contemplated application. The relaxation of weight, volume, and power constraints on spacecraft design will permit designers to consider a much wider range of qualified equipment for a given application, and to select equipment that historically would have been rejected as over-designed, oversize, or overweight. Much more emphasis must be placed on avoiding RDT&E costs in future space programs. Minimize new technology and hardware development unless (1) there is a mandatory new requirement and (2) it has been proven cost-effective relative to available technology and hardware.

6.2.4 Low-Density Electronic Packaging

Electronic equipment for space programs has evolved in the direction of reduced size and weight, and increased packaging density. In general, this evolution has been accompanied by cost escalation. The relaxation of weight and volume constraints in the Shuttle era should permit the development of simpler, lower density packaging techniques for electronic equipment; resulting in the reduction of RDT&E and recurring costs of such equipment. Low density electronic packaging saves labor costs in design, modification, fabrication and assembly, repair, and inspection. Provide volume for low-density equipment installations to simplify installation and insure complete accessibility of parts and components for inspection, test, or replacement.

6.2.5 Over-Designed Structure

The payload weight and volume capability of the Shuttle will permit the design of very conservative payload structure with factors of safety high enough to eliminate or reduce requirements for detailed structural load testing. This over-design approach also will simplify the analysis of special and redundant failure modes such as shell-buckling; ruggedization of structural elements will force most failures into simple failure modes. Use high factors of safety (three or greater) for sizing structural elements. Reduce design analysis and testing requirements for structures.

6.2.6 Structures for Minimum Alignment-Tooling

The costs of tooling for space payloads are directly related to the tolerances specified for the assembly and alignment of structures and for the installation of equipment. The larger, heavier structures of Shuttle-launched payloads should not require as much tooling as the lighter, more flexible structures of historical payloads. Also, tooling costs can be reduced if, in the design of a payload structure, a single structurally-stable plane can be established on which all equipment requiring critical alignment can be mounted. Increase dimensional tolerances.

6.2.7 Elimination of Weight-Control Processes

Manufacturing costs can be reduced if payload designers avoid sandwich materials, chemical milling, machine contouring, and other techniques for weight control commonly used in the design of payloads for expendable boosters but unnecessary in the design of payloads for the Shuttle. Minimize use of contour-machined parts and eliminate close-tolerance machining for weight reduction. Minimize use of sandwich materials. Avoid use of higher-cost parts such as castings or forgings.

6.2.8 Low-Density Modularization for Manufacturing/Test

Low-density packaging of equipment into modules and of modules into payload structures will help to reduce manufacturing and testing costs by providing:

- Free access to components within each module
- Complete bench test of each module or combination thereof prior to spaceframe installation
- Quick installation of module into spaceframe or removal for failure repair, inspection.

6.2.9 Modularization of Spacecraft Equipment for In-Orbit Replacement

In the design of spacecraft for launch by the Space Shuttle, equipment installations should be modularized to permit in-orbit removal and replacement. The capability to repair a spacecraft in the Shuttle cargo bay* before releasing it is a major factor in the reduction of space program costs that will be realized through the use of the Shuttle. The following guidelines are applicable to the design of equipment modules for Shuttle payloads:

- Divide payload subsystems into minimum quantity of modules consistent with:
    - Maximum weight/size which can be readily installed or removed by a shuttle crewman
    - Maximum cost of a single module (dependent on type/quantity of components therein) should allow repair/replacement at a reasonably low percentage of total spacecraft cost.

* or on extended deployment mechanisms

- Segregate components into logical module groupings with similar functional requirements, predicted life.
- Establish operating tolerances on individual modules so that module replacement will not require payload recalibration.
- Provide simple functional and mechanical interfaces between module, spaceframe, and other modules.
- Provide for easy access to and removal/installation of modules without need for special tools by astronaut or by automated teleoperator or manipulator.

6.2.10 Elimination or Simplification of Mechanisms

Use of the Shuttle will allow elimination of some deployment mechanisms and simplification of others. On smaller payloads, solar arrays, antennas, and other devices can be mounted in the space-deployed configuration. On other (larger) installations, the degree of folding for launch/ascent stowage can be reduced and the space-deployment device simplified (the delatching and extension can be aided by the Shuttle manipulator (as a backup) if desired.

Eliminate deployment mechanisms when payload space envelope permits fixed installation of solar panels, antennas, sensors, and other equipment.

Avoid sophistication and miniaturization of mechanisms. Avoid mechanisms which are not self-supporting in one-g.

6.2.11 Thermal Control Simplifications

With the average size of future payloads increasing, the amount of surface area available for thermal control will become significantly larger. The use of low-cost passive thermal control will be more widely used. Also, the volume-limit relaxation will allow separate thermal isolation of experiment packages, simplifying the thermal control of both the experiment and the spacecraft. Isolate experiment thermal control. Increase thermal operating range where possible. Maximize use of passive thermal control.

6.3 Considerations for Low-Cost Ground Support Equipment

There are a number of often overlooked things the payload designer can do to minimize the costs of GSE and the recurring operations associated with handling, transportation, checkout, storage and logistics support. If such considerations are incorporated at the outset of the program they do not add to the design burden, eliminate the necessity to modify designs later to incorporate support provisions, reduce the amount of new GSE that must be designed, and minimize the amount of STE and tooling needed to produce the payload. Some of the factors listed below are system criteria, some relate to structural and mechanical design, and others impact electrical and/or software design; they are all important in minimizing total payload program cost.

6.3.1 Ruggedized Payload Effect on GSE

The design requirements for structures, subassemblies, components, and piece parts for low cost payloads should include the ground handling and transportation loads; this is contrary to some past practices where weight limitations dictated that ground conditions should not place limitations upon the design. However, the cumulative effect of special handling, special testing, special ground support equipment, and all the associated procedures, validations and documentation is higher development and operating costs.

These costs can be minimized under the "low-cost" payload guidelines. Simple design trade studies that consider the cost of GSE will usually provide a quick indication of whether to "beef up" the payload or otherwise increase the capability of the payload or to require additional complexity of ground operations and GSE.

For example, maximum Shuttle-imposed loads are 3.3g, whereas typical in-plant dollies probably do not impose loads of more than ±2g vertically and ±1g in the horizontal plane. However, if shipment of the payload or components by truck or by air is necessary, loads of ±6g may be encountered. Therefore, a component such as a solar array panel (stowed position) should be designed to be self-supporting for simplicity in assembly, inspection, and testing, and to withstand ±6g ground transportation loads. If the array can be simply removed for shipping, this approach would not apply.

Shipping and storage containers will still be necessary to protect the components from contaminants and to support them at the designed load bearing points, but complex shock mitigation systems and recording equipment, and special handling are not required.

Similar treatment should be given to other payload elements such as antennas, main structure, modules and large experiment structure or packages. It should be noted that choice of materials, finishes and assembly techniques are as important in the ground environment as the structural load factors, and the term "ruggedized" applies to the total environment.

6.3.2 Hard Points and Lift Provisions

Suitable mounting provisions and load paths are designed into payloads to mate with booster adapters or specified primary interfaces, but often the need to hoist, tilt, rotate and translate in order to fabricate, assemble, test, ship, store and mate is neglected. In the past this has resulted in the design of expensive handling fixtures, complex shipping containers and tricky mating maneuvers.

The removal of severe payload weight and volume constraints allows the incorporation of lifting pads or hard points, lifting eyes or holes, handholds, and tiedown brackets, holes, recesses or cleats. Such provisions should be provided for the assembled spacecraft and independently for each interchangeable module and replaceable component as appropriate to its size and weight.

6.3.3 Alignment References

It will often be desirable to determine the principal axes of payloads for checkout as well as installation into Shuttle. External optical reference marks or targets allow this to be done quickly with common optical instruments, and reverified as often as necessary. Where the reference is transferred from a plug-in module, a target on the module may also be desirable. The targets should be located so they are visible through the cargo bay doors and/or access ports. Each payload module should be self-aligning to the payload main structure to a specified accuracy, and for any position or attitude of the payload.

6.3.4 Thermal Design of the Payload

The thermal design is usually optimized for the space environment, but the ground checkout and pre-launch environment must not be neglected. GSE can supply dry, conditioned air or a nitrogen purge to a general area of the payload, but it requires equipment and personnel that must be maintained and trained, slows up operations and is usually avoidable if consideration has been given to thermal paths and radiative surfaces. While power-on cycles can be limited under ground conditions such limitations slow up testing, increase integration complexity and are, in general, undesirable and within the designer's prerogative to control.

6.3.5 Protection of the Payload

All appendages, alignment devices, connectors, fasteners, sensors, etc., should be protected against handling damage, preferably by placement on or within primary structure, or, alternatively, by simple fenders or guards. Protective caps or covers should be used for optics, connectors and surfaces sensitive to dust, moisture, and other contaminants; these should be designed for ready accessibility for removal during test or launch operations. Wherever possible systems and materials that do not require such protection should be selected.

6.3.6 Checkout of Payload

Testing, from the piece-part level through the payload level, represents a very significant portion of the payload costs; during manufacturing, assembly, final acceptance testing, and pre-launch and in-flight checkout. As such, it represents a very fruitful area for cost reduction. Specific examples of low-cost payload design for reduction in test/checkout requirements are: (1) enlarged tolerances in operating parameters, large factors of safety, or enlarged performance margins; (2) built-in test provisions for each "black box"; (3) separate functional verification of redundant elements; (4) interchangeability of replaceable modules/"black boxes" without recalibration; and (5) independent safety-monitoring instrumentation.

In addition, ease of orbital checkout, enabling a quick verification of payload functional capability, will reduce Shuttle "stay time" and therefore reduce costs, as well as permit repair of launch/ascent-induced problems. Thus, payload systems should be configured to utilize automatic, computer-controlled payload test sets (PTS), should such be adopted, which will provide a standard power, command, and data-processing interface with the Shuttle.

6.4 Consideration for Low-Cost Launch and Flight Operations

Recurring operations, particularly on programs having long operational life, account for an appreciable fraction of the total program cost; this is principally evident in the manning level required. Systems designed to require less support, simpler procedures and fewer activities or operations will cost less. Mandatory ground rules should include use of: (1) common support equipment; (2) standard RF links and data formats; (3) shared control facilities; (4) STADAN and NASCOM services, including TDRS when available; and (5) existing government data processing capabilities.

6.4.1 Effect of Safety Factors

All pressure vessels, tanks, pyrotechnics, radiation sources, or toxic materials used in low-cost payloads must be self-safing, inherently safe as a result of the safety factors used in the design, or capable of being rendered safe by remote command. It is desirable that pre-flight servicing, including loading of propellants and pressurants, be done quickly and without the necessity of clearing the area of personnel or requiring that other activities in the vicinity be suspended. Safe-Arm plugs or switches should be provided to minimize hazards to launch vehicles and personnel during the pre-countdown joint flight acceptance compatibility tests.

6.4.2 Test Connectors

Means should be provided for conducting RF closed-loop tests without having to obtain range clearance to radiate. Safety instrumentation sensor outputs should be routed directly to test connectors so checks can be made without having to activate payload power, signal conditioning and data subsystems. Hardline controls should be provided for safing and initialization. All test connectors should be designed fail-safe for disconnect.

6.4.3 Software

The payload system software should be compatible with payload test set interfaces and should be designed for ease of programming changes in test and operations procedures. The need for computer programmers in routine operations modes should be avoided. Extensive use of independent subroutines for specific operations is desired, with internal program controls to prevent inadvertent memory dumps or initiation of hazardous sequences. It is important that the software system design be established concurrently with hardware design requirement formulation. On-board memory capacity should be at least twice the estimated basic requirement so that straightforward programming techniques can be used and real-time changes can be made without having to employ sophisticated memory conservation devices.

6.4.4 Autonomous Orbital Operations of the Payload

Autonomy of the payload operation can be a most effective design feature for reducing launch and flight operations costs, both direct charges to the program, and the services supplied by other government-supported activities. An autonomous payload should be able to maintain its health (i.e., power, attitude, temperature, configuration and sensor protection) for long periods without ground attendance, including substitution of backup equipment or functions to correct failure modes. This implies a self-check capability, a large stored command repertoire, and a passive "storage" mode. The degree to which these capabilities are incorporated in any particular payload should be decided by a tradeoff among: (1) the costs for providing the capability; (2) the costs to perform the repair functions (replacement modules and Shuttle/Tug flight); (3) the comparative costs of ground tracking and data acquisition system. Ancillary benefits should also be considered: more freedom for scheduling of ground network resources; more flexibility in planning payload operations and modifying same; potential of sharing sustaining engineering support because subsystem specialists and computer programmer support will generally not be needed for updates/mods to payload on-board computer program.

Section 7
IMPLEMENTATION OF STANDARDIZATION FOR FUTURE SPACECRAFT

With the Shuttle as a forcing function toward commonalization of missions and orbits, the historically costly mission-peculiar approach can be replaced with hardware standardization and the corollary significant cost savings. This section is devoted to explaining generalized standard hardware implementation, using actual examples from recently-completed designs and analyses.

7.1 Design of Standard Subsystems and Modules

7.1.1 General Approach to Standard Subsystem Design

The first major step in standard hardware implementation is the preparation of conceptual designs for standard spacecraft subsystems for application to mission-peculiar spacecraft, Standard Spacecraft, and Cluster Spacecraft. The subsystems selected for standardization should be those which in general have universal application to most of the future missions; samples are outlined in the following paragraphs. The Structures subsystems and the Environmental Control subsystems, because they vary widely with specific mission requirements, are not considered for standardization initially. Also, because there is limited application of a Propulsion subsystem in the group of unmanned satellites considered in the sample (NASA unmanned missions) only a modest effort was applied to its standardization.

7.1.1.1 Stabilization and Control (S&C).

Function: To orient and stabilize the spacecraft

Equipment: Sensors, inertial reference units, reaction or momentum wheels, magnetic torquers, gyros, and control electronics

7.1.1.2 Communication, Data Processing and Instrumentation (CDPI).

Function: To obtain spacecraft status data, process spacecraft and experiment data and commands, perform computation and timing functions, and provide communication with other spacecraft, the Shuttle, or with ground stations

Equipment: Sensors, signal conditioners and multiplexers, digital computers and ancillary equipment, receivers, transmitters, and antennas.

7.1.1.3 Electrical Power (EPS).

Function: To generate, store, condition and distribute electrical power, and to distribute electrical signals.

Equipment: Solar arrays, charge controllers, batteries, power regulators and converters, power control and distribution equipment, and electrical harnesses.

7.1.1.4 Attitude Control (ACS).

Function: To provide thrust for torquing and translating the spacecraft.

Equipment: Propellant tanks, control valves, plumbing, and thrusters.

7.1.2 Basic Design Criteria

The following basic design criteria should be applied in the conceptual design of standard subsystems:

- Optimize for low program cost (low-cost payload design guidelines)
- Utilize modular-packaged equipment to allow internal modifications without changing module interfaces
- Provide for on-orbit replacement of equipment modules with maximum accessibility
- Provide for equipment module replacement without need for spacecraft recalibration
- Provide for minimum on-orbit checkout utilizing Shuttle on-board checkout

- Provide simple interfaces with Shuttle systems
- Eliminate deployment mechanisms or provide for deployment prior to orbit-release from Shuttle (antenna, sensors, solar arrays, etc.)
- Provide for growth and update in all module packaging and at all interfaces
- Provide multi-mission interface compatibility for experiment packages
- Provide for space docking and orbit retrieval by Shuttle, Tug, or Teleoperator
- Provide for man-safety (not man-rating)

7.1.3 Mission Equipment Support Requirements

To establish design requirements for standard subsystems, it is necessary to (1) Establish the basic orbit parameters for each mission/spacecraft, and (2) ascertain the support requirements of the mission equipment (experiment sensors, transponders, etc.) of each of the missions in the pertinent Mission Model.

Figure 7-1 illustrates a sample of the orbit parameter listing, showing altitude and inclination, quantity of spacecraft for total mission duration and quantity

Fleming No.	Mission		Altitude		Incl. (deg)	Total Quantity of S/C	Qty. of S/C in Orbit Set
	Cat.	Title	Km	(NM)			
1	Physics/Astronomy	Astronomy Explorer A	500	(270)	28.5	15	1
3		Magnetosphere - Low	3520/260	(1900/140)	28.5 - 90	12	1
6		OSO	650	(350)	<35	1	1
7		Gravity/Relativity	930	(500)	90	2	1
13		H.E. Astronomical Obs.	650	(350)	28.5	6	1
15		Large Stellar Telescope	650	(350)	28.5	5	1
17		Large Solar Observatory	650	(350)	30	4	1
19		Large Radio Observatory	650	(350)	30	3	1
21	Earth Observation	Polar EOS	930	(500)	99 SS*	12	4
23		Earth Physics	740	(400)	90	7	1
25		TIROS	1300	(700)	101 SS	3	1
26		Polar ERS	930	(500)	99 SS	6	1 - 6
30	Com/Nav	Small ATS	5550/555	(3000/300)	0 - 90	12	1
32		Co-op ATS	5550/555	(3000/300)	0 - 90	2	1
75	Non-NASA	TOS Met	1300	(700)	101 SS	12	1 - 3
77		Polar ERS	930	(500)	99 SS	22	3/3

* SS - Sun Synch.
** Without Adapter.

Fig. 7-1 Baseline LEO Mission Parameters - 1979-1990 (Sample) of spacecraft in orbit at any unit of time. The required design life of each spacecraft and its experiments/sensors should also be made a part of the reference data package.

Figure 7-2 is a sample data sheet which lists the characteristics of each experiment/sensor for specific missions and the corollary spacecraft support requirements. These data must be carefully analyzed in establishing performance requirements for standard subsystems and in their assignment to each of the mission-peculiar spacecraft.

7.1.4 Standard Subsystems Characteristics and Equipment

Point designs of typical standard subsystems should be developed next as a base for deriving complements of components, weight and volume estimates, and cost estimates. In the LMSC Payload Effects Follow-On study, point designs of typical standard subsystems have been documented in LMSC Engineering Memos, which are listed in the following table and are included in Vol. II of this Design Guide.

LMSC Engineering Memos					
Mission	Spacecraft General Description	Standard Subsystem			
		S&C	CDPI	EPS	ACS
Earth Observatory Satellite	PE-106	PE-102	PE-103	PE-104	PE-105
Communication Satellite	PE-126	PE-122	PE-123	PE-124	PE-125
Planetary Satellite			PE-133		

The general characteristics and major equipment of these standard subsystems, as developed separately for a 1-year Earth Observatory Satellite and a 5-year Communications Satellite are summarized in Figs. 7-3 through 7-6.

MISSION EQUIPMENT			No. Req	Aperture or FOV	RESOLUTION		Spectral Bands	Unit Size (Ft³)	Unit Weight (lbs)	POWER		POINTING					DATA OUT		
No.	Name	Objective			Ground Area	Altitude				Average	Duty Cycle	Direction	Accuracy	Stability	Attitude Rate	Method	A or D	BPS (max)	Duty Cycle
1	Astronomy Explorer	Solar and Stellar Investigations	-	-	-	-	-	200	200	50	.3	Stars, Sun	-	-	-	Vehicle to ± 15'	-	-	-
	UV Telescope Assy	UV Spectrums & Imaging	1	18"	0.3 arc sec		.12-.30 µM	10	170	25	.3	-	10"	10"	-	Tel-Gimb. & Int. Corr.	D	3 MBPS	.3
	Proportional Counter	Catalog position, strength Spect.Conf.of X-Ray Sources	2	1°	-	-	2-20 KEV	5	15	3	1.00	-	6'	12"	-	-	D	4.6 KBPS	1.00
2	Radio Astronomy Explorer	Meas. Stellar and Planetary Intensity & Spectra	-	30 ft	-	-	50 Kc - 100 Mc	-	250	50	1.00	Stars, Planets			1° sec	Vehicle to ± 15'	-	-	
	Detector Assy	Receive ELF, VLF, LF, UHF, VHF and HF Star Signal	1	-	-	-	Visual	2.0	150	50	1.00	-	-	-	-	Tel.-Gimb. & Int. Corr.	D	1 KBPS	1.00
3	Lower Magnetosphere Explorer	Investigate Lower Magnetosphere	-	-	-	-	-	-	100	100	1.00	Earth	2° (Spin)	-	-	N/A	-	-	-
	UV Radiometer	Measure Background UV Radiation	1	4"	-	-	0.1 - .4 µM	5	8	10	1.00			-	-		D	8 MBPS	-
	VHF Radio Receiver	Measure Background Radiation	1	2M	-	-	50 Kc- 1 Mc	10	36	50	1.00			-	-		A	90 KHZ	1.00
	Magnetometer	Investigate Earth's Field	1	-	-	-	-	6	7.5	7.5	1.00			-	-		A	10 HZ	1.00
	Mass Spectrometer	Meas. Ion Composit. of Incoming Particles	1	40° FOV	-	-	-	7.5	26	14	1.00			-	-		D	810	1.00
	Mass Spectrometer	Molecular Analysis	1	40° FOV	-	-	-	2	22	20	1.00			-	-		D	540	1.00
4	Middle Magnetosphere Explorer	Investigate Middle Magnetosphere	-	-	-	-	-	-	100	30	1.00	Earth	2° (Spin)	-	-		-	-	-
	Langmuir Probe	Meas. Electron Density & Body Potent.	1	-	-	-	-	.04	2	3.3	1.00			-	-		A	126 cps	1.00
	Magnetometer	Investigate Earth's Magnetic Field	1	40° FOV	-	-	-	.5	7.5	7.5	1.00			-	-		A	10 HZ	1.00
	Mass Spectrometer	Meas. Ion Composition of Incoming Particles	1	40° FOV	-	-	-	3	50	10	1.00			-	-		D	810	1.00
	Drag	Meas. Atmospheric Drag	1	Large	-	-	-	Large	35	5	1.00			-	-		A	10 HZ	1.00
5	Upper Magnetosphere Explorer	Investigate Upper Magnetosphere	-	-	-	-	-	-	150	40	1.00	Sun	2° (Spin)	-	-		-	-	-
	Magnetometer	Investigate earth's Magnetic Field	1	-	-	-	-	.5	7.5	7.5	1.00			-	-		A	10HZ	1.00
	Mass Spectrometer	Measure Ion Comp. of Incoming Particles	1	40° FOV	-	-	-	.3	50	10	1.00			-	-		D	810	1.00
	Solar Wind Detector	Measure Solar Wind Intensity	1	Large	-	-	-	Large	35	5	1.00			-	-		A	10HZ	1.00
	Mass Spectrometer	Molecular Analysis	1	40° FOV	-	-	-	2	22	20	1.00			-	-		D	540	1.00
6	Orbiting Solar Obs.	Measure Temporal Variation of Sun's Bright.	-	-	-	-	-	-	500	150	.5	Sun	± 5"	-	-	Vehicle to ±30"	-	-	
	U.V. Telescope Assy.	Spectroscopy, Photometry, and Imaging of Sun in U.V.	1	.25M	1 arc sec		.1 - .4 µM	29	30	35	.5		±10"	2.5"	-	Tel-Int. Corr.	D	8 MBPS	.50
	X-Ray Telescope Assy.	Spectroscopy, Imaging, and Meas. of Sun in X-Ray	1	.1M	1 arc sec		2×10^{-4} to 10^{-2} µM	22	130	55	.5		±5"	2.5"	-		D	119 MBPS	.50
	UV Spectroheliograph	Meas. Temp. Var. of Sun Brightness	1	.125M	1 arc sec		.1 - .4 µM	25	140	32	.5		±10"	2.5"			D	175 MBPS	.50
	X-Ray Spectroheliograph	Meas. Temp. Var. of Sun Brightness	1	.125M	1 arc sec		2×10^{-4} to 10^{-2} µM	17	150	36	.5		±10"	2.5"			D	175 MBPS	.50

Fig. 7-2 Typical Listing of Experiment/Sensor Characteristics and Spacecraft Support Requirements

Earth Observatory Satellite	Communication Satellite
Characteristics: • Earth-oriented; one-year life • ± 0.5-deg attitude pointing, ±0.006 deg attitude determination (3σ) • No earth-sensing provided • Attitude rate control to ±0.005 deg/sec • Magnetic torque wheel unloading • Mass expulsion used only for backup attitude hold Major Equipment: • 2 Fixed-head star trackers • Three-axis precision rate sensor (redundant) • On-board attitude and attitude control computations in CDPI computer • Three single-axis reaction wheels • Three single-axis magnetic torquers • 40-50m ephemeris every 20 min via TDRS • Sun sensors + rate gyros for backup modes	Characteristics: • Earth-oriented; five-year life • ±0.16-deg narrow beam pointing accuracy (3σ) • 100 ft-lb-sec pitch momentum bias • No yaw sensing required except during North-South stationkeeping • Mass expulsion for stationkeeping, wheel unloading and for backup attitude hold Major Equipment: • Long-life earth horizon sensor (redundant) • Dual gimbal twin pitch momentum wheels • On-board attitude control computations in CDPI software • Solar Aspect Sensors for yaw attitude during stationkeeping • Same sun sensors + rate gyros for backup

Fig. 7-3 Features of Typical Standard Subsystem - Stabilization & Control

Earth Observatory Satellite	Communication Satellite
Characteristics: • Communications via TDRS system • No on-board Mass Data Storage • On-Board Computer Control • Command Control via Link • CDPI components included in Mission Equipment Data Path Major Equipment: • Communication Section: K-Band Transmitter & Receiver S-Band Transmitter & Receiver VHF Transmitter & Receiver Ranger Antennas-Gimballed & Omni • Interface Section: Ultra High Data Rate Unit - logic, registers High Data Rate Unit - logic, registers, counters Low Data Rate Unit - logic, registers, counters, timer A to D converters, multiplexers • Data Processing Section: Digital Computer - 4th generation, 16K memory • Instrumentation Section: Analog, Digital, Bi-level Transducers	Characteristics: • Communication to Ground Stations • No On-Board Mass Data Storage • On-Board Computer Control • Command Control via Link • CDPI Components not in Mission Equipment Data Path Major Equipment: • Communication Section: S-Band Transmitter & Receiver Ranger Antennas - Omni • Interface Section: Low Data Rate Unit - Logic, registers, counters, timer, A to D converters, multiplexers • Data Processing Section: Digital Computer - 4th generation, 16K memory • Instrumentation Section: Analog, Digital, Bi-level Transducers

Fig. 7-4 Features of Typical Standard Subsystem - CDPI

Earth Observatory Satellite	Communication Satellite
Characteristics: • 1000 watts ave. - end of life • Fixed solar array, pre-launch beta adjust • Unreg. Bus: 25 to 28 VDC • Regulated Bus: 28 ± 2% VDC • Array Switching	Characteristics: • 1750 watts ave. - end of life • Tracking array - single axis • Unreg. Bus: 25 to 28 VDC • Regulated Bus: 28 ± 2% VDC • No Array Switching
Major Equipment: • 380 sq ft Solar Array • 6 40 amp-hr batteries • 6 Charge Controllers • 1 DC-DC Regulator	Major Equipment: • 263 sq ft Solar Array • 4 40 amp-hr batteries • 4 Charge Controllers • 1 DC-DC Regulator • Solar Array Drive Assy • Solar Array Regulator

Fig. 7-5 Features of Typical Standard Subsystem - Electrical Power

Earth Observatory Satellite	Communication Satellite
Characteristics: • Freon 14 Propellant • Provides control with any 3 of 4 modules • Qualified hardware • Module wet weight = 140 lbs • Module size 22" x 32" x 24" • 2 yr orbital life	Characteristics: • Hydrazine Monopropellant • Simple Blowdown Feed System • Dual Series Thruster Valves for Leakage Redundancy • Module wet weight = 267 lbs • Module size 28" x 28" x 44" • Provides control with any Single Thruster Failure • 5 yr orbital life
Major Equipment: • 16" D. stainless steel storage tank • Pressure Regulator & Solenoid Valve Assy. • 4 Clusters of four 1.75 lb thrusters • Fill valve	Major Equipment: • 27" D. stainless steel storage tank • Propellant management screen inside tank • 4 Clusters of six 0.5 lb thrusters • Fill valve

Fig. 7-6 Features of Typical Standard Subsystem - Attitude Control

7.1.5 Typical Standard Modules

The equipment listed for each subsystem are then grouped into modules in acccordance with the modularization guidelines presented in par. 6.2.9. Typical modules representing the subsystems of an Earth Observatory Satellite are described in Fig. 7-7.

Subsystem	Module	Equipment in Module	Module Weight (lb)
Stabilization & Control	Primary Sensing Module No. 1	• Fixed Head Star Trackers (2) • FHST Electronics (2) • Three-Axis Rate Sensor • Precision Equipment Mount • Module Base • Module Cover • Cables and Connectors	Basic 91 lb 15% contingency 14 Total 105 lb
Stabilization & Control	Secondary Sensing Module No. 1	• Sun Aspect Sensor (5) • Sun Aspect Sensor Electronics • Rate Gyro Package • Secondary Stabilization & Control Electronics • Module Base • Module Cover • Cables & Connectors	Basic 56 lbs 15% contingency 9 Total 64 lbs
Stabilization & Control	Reaction Torque Module No. 1	• Reaction Wheel (3) • Wheel Support and Safety Shield • Wheel Drive Electronics • Magnetic Torquer (3) • Mag. Torquer Electronics (3) • Module Base • Module Cover • Cables & Connectors	Basic 133 lbs 15% contingency 20 Total 153 lb
Communication Data Processing & Instrumentation	K_u-Band Communication Module No. 1	• K-band TWTA (50 watts out) (2) • K-band PLL Receiver • K-band QPSK Modulator/Driver • K-band Multicoupler • Interface Unit (High Rate) • Module Base • Module Cover • Waveguide, Cables, Connectors	Basic 74 lbs 15% contingency 11 Total 85 lbs

Fig. 7-7 Typical Earth Observatory Satellite Subsystem Modules (Partial List Only)

A module dimension, preferably the same size for all modules is then selected. Each module-set of components is then arranged within the selected volume; volume overage is provided for later additions or changes to the initial module components.

A typical module configuration is shown in Fig. 7-8.

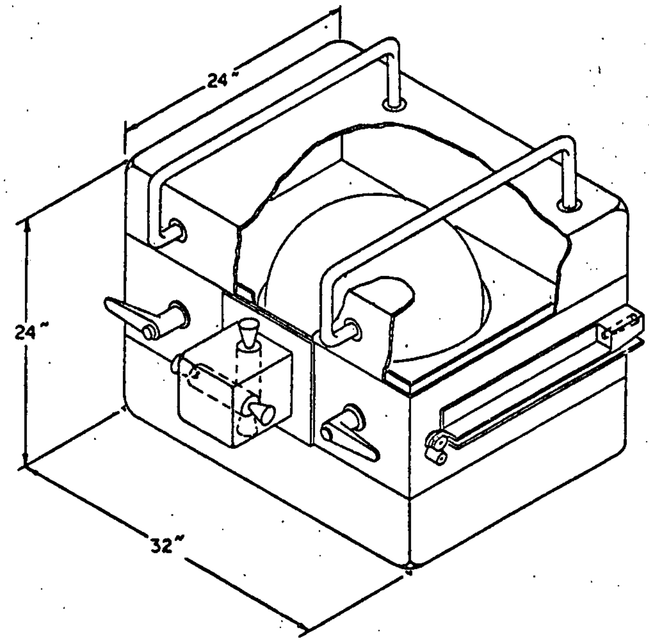


Fig. 7-8 Attitude Control Subsystem-Typical Standard Module

Each module is designed to be guided into its location in the spacecraft by rails and aligned and supported by two inboard pins and two outboard cams that engage machined grooves in the rails. The cams also transmit force from the cam actuators on the outboard face of the module to accomplish the controlled engagement and disengagement of the bulkhead-type electrical connectors on the in-board face of the module. The two wrap-around handles are designed to facilitate the handling of the module in orbit by a Space Shuttle crewman.

7.1.6 Typical Spacecraft Incorporating Standard Modules

To assure that the modules can be arranged in a satisfactory manner within a spaceframe, considering module-to-module compatibility, experiment/sensor locations, etc., a spacecraft layout should be made. Conceptual integrated designs of an Earth Observatory Satellite and a Communication Satellite incorporating standard subsystem modules are described briefly in the following paragraphs.

7.1.6.1 Earth Observatory Satellite (Future Version for Shuttle Era). The Earth Observatory Satellite (EOS) is to be launched by the Space Shuttle and is designed to be checked out and repaired, if necessary, in the Shuttle prior to being placed into the mission orbit; to be repaired in orbit by the Shuttle during its design lifetime of one year; and to be recovered from orbit by the Shuttle after one year or longer for complete refurbishment and subsequent return to orbit. All communication with the EOS is assumed to be via a Tracking and Data Relay Satellite system of three equally-spaced synchronous equatorial satellites. No on-board data storage is provided in the EOS design.

The nominal orbit of the EOS is near-polar circular and sun-synchronous, with altitude = 485 nm and inclination = 97 degrees. The number of orbits per day is 14.

The mission of the EOS is as follows:

- Provide a facility for the conduct of experimental research and development of advanced space systems for the earth observations disciplines.
- Obtain data in both the visible and infrared spectral bands to detect and distinguish the signatures of agricultural and forest resources and of natural thermal sources.
- Perform space observations of oceanographic phenomena and interactions of the ocean surface with the atmosphere to meet urgent needs for research data and the development of advanced operational sensors for the oceanographic and meteorological disciplines.
- Initiate a program to develop and test space sensors to monitor indicators of environmental quality, such as atmospheric pollution, on global and other appropriate scales.
- Provide a flexible data management system having the capability of providing data in appropriate formats and quantities on a timely basis, primarily for research purposes but also (as may be indicated during the initial research phase of one or more of the instruments) for quasi-operational use by appropriate agencies on a real-time basis.
- Develop a low-cost spacecraft for launch by the Space Shuttle, adaptable to supporting a wide variety of earth observations sensors and combinations of sensors.

The general configuration of a future EOS is shown in Fig. 7-9 and the location of equipment in Fig. 7-10. The spacecraft subsystem equipment has been packaged in modules that can be removed and replaced in orbit by a Space Shuttle crewman

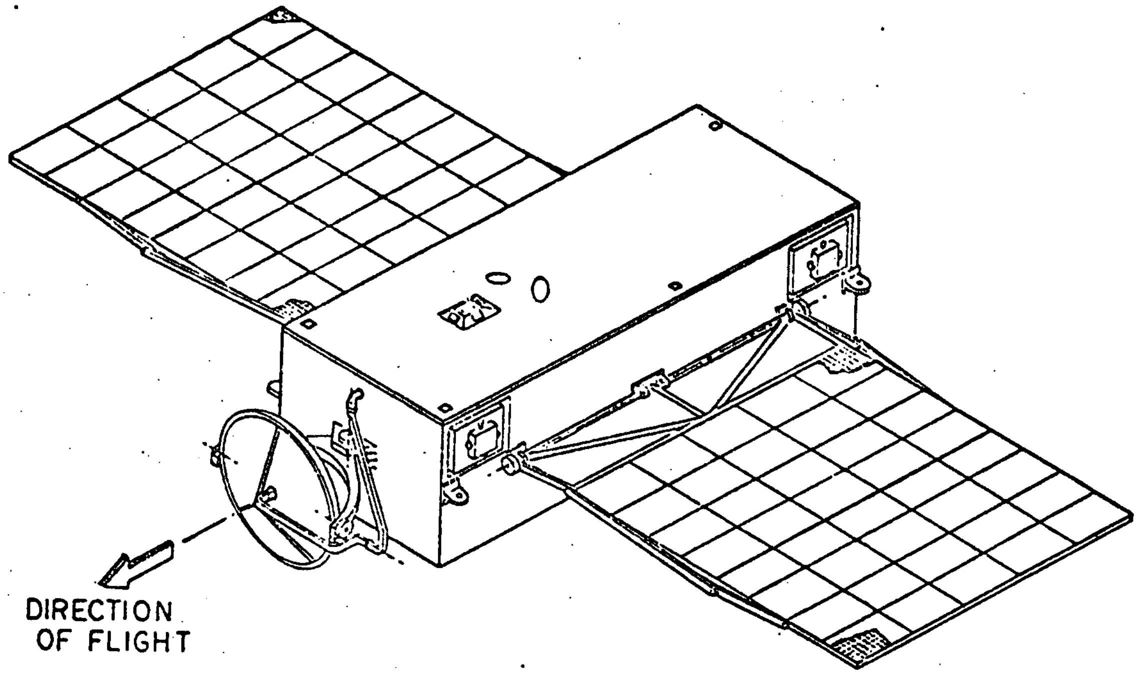


Fig. 7-9 General Configuration - Future Earth Observatory Satellite

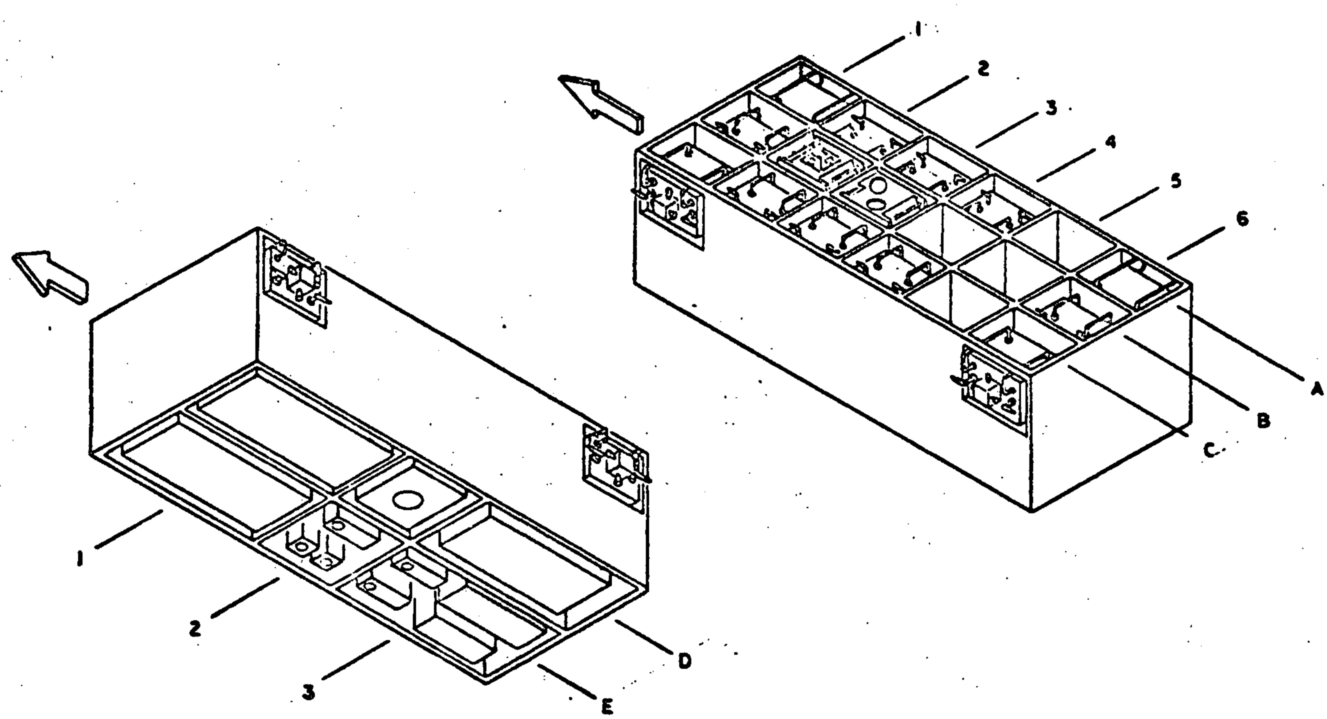


Fig. 7-10 Equipment Locations - Earth Observatory Satellite

or by an automated Teleoperator or manipulator. This makes possible the check-out and rapid repair, if necessary, of the EOS in orbit if it should fail prior to recovery for refurbishment. The mission equipment (the seven sensors) can be packaged in similar replaceable modules. The specific identification of modules (correlating with letter-number matrix on Fig. 7-10) is shown on Fig. 7-11.

The EOS configuration provides spare volume for growth of either spacecraft subsystems (see Compartments A-5, B-4, B-5, and C-5 on Fig. 7-10) or mission equipment.

Mission Equipment	Spacecraft Subsystem Modules
D-1 Passive Microwave Radiometer (λ = 0.81 cm)	A-1 Attitude Control Module No. 1
	A-2 S & VHF Band Communication Mod
D-2 Thematic Mapper	A-3 Battery Module No. 1
D-3 Passive Microwave Radiometer (λ = 2.81 cm)	A-4 Power Control Module
	A-5 Empty
E-1 Passive Microwave Radiometer (λ = 6.01 cm)	A-6 Attitude Control Module No. 2
	B-1 K-Band Communication Module
E-2 Ocean Scanning Spectrophotometer Atmospheric Pollution Sensor Upper Atmosphere Sounder	B-2 S&C Secondary Reference Module
	B-3 S&C Primary Reference Module
	B-4 Empty
E-3 Cloud Physics Radiometer Sea Surface Temp. Radiometer Passive MW Radiometer (λ = 1.67 cm) Passive MW Radiometer (λ = 1.40 cm)	B-5 Empty
	B-6 Reaction Torque Module
	C-1 Attitude Control Module No. 3
	C-2 Data Processing Module
	C-3 Battery Module No. 2
	C-4 Battery Module No. 3
	C-5 Empty
	C-6 Attitude Control Module No. 4

Fig. 7-11 Equipment Module Identification - Earth Observatory Satellite

7.1.6.2 Communications Satellite (Future Version for Shuttle Era). The Communication Satellite is to be placed into geosynchronous orbit by the Space Shuttle and Space Tug. The satellite and the Space Tug are mated at the launch base and installed in the cargo bay of the Space Shuttle, which is then launched into a low-earth parking orbit. The satellite is designed to be checked out in the Shuttle and repaired, if necessary, by the replacement of equipment modules prior to being transported to geosynchronous orbit by the Space Tug. The satellite may be recovered from its operational orbit and returned to earth for repair, refurbishment and reuse; or it may be repaired in geosynchronous orbit by a Teleoperator if a Space Tug/Teleoperator system is developed.

The general configuration of a future Communication Satellite in flight is shown in Fig. 7-12. The direction of flight is eastward and the single-axis-tracking solar power panels are extended to the north and south of the spacecraft.

The location of equipment modules is shown in Fig. 7-13. The modules are designed to be readily removed and replaced in orbit by a Space Shuttle crewman or by automated Teleoperator/manipulator.

The modules in locations B-1, B-3, D-1, and D-3 are protected from solar radiation by hinged doors. The exposed surfaces of the remaining modules will be protected by appropriate surface finishes and insulation as determined by detail analysis of thermal control requirements.

7.1.7 Standard Subsystem Modules for Mission Model

The group of subsystem modules designed for the point-design satellites (Earth Observatory Satellite and the Communication Satellite) must be augmented by additional modules to establish an inventory of standard modules to accommodate the various missions in the Mission Model.

7.1.7.1 Screening Out Very Special Missions. Detail inspection of the subsystem requirements for the various missions is necessary to screen out those mission applications which have very special requirements. An example is the

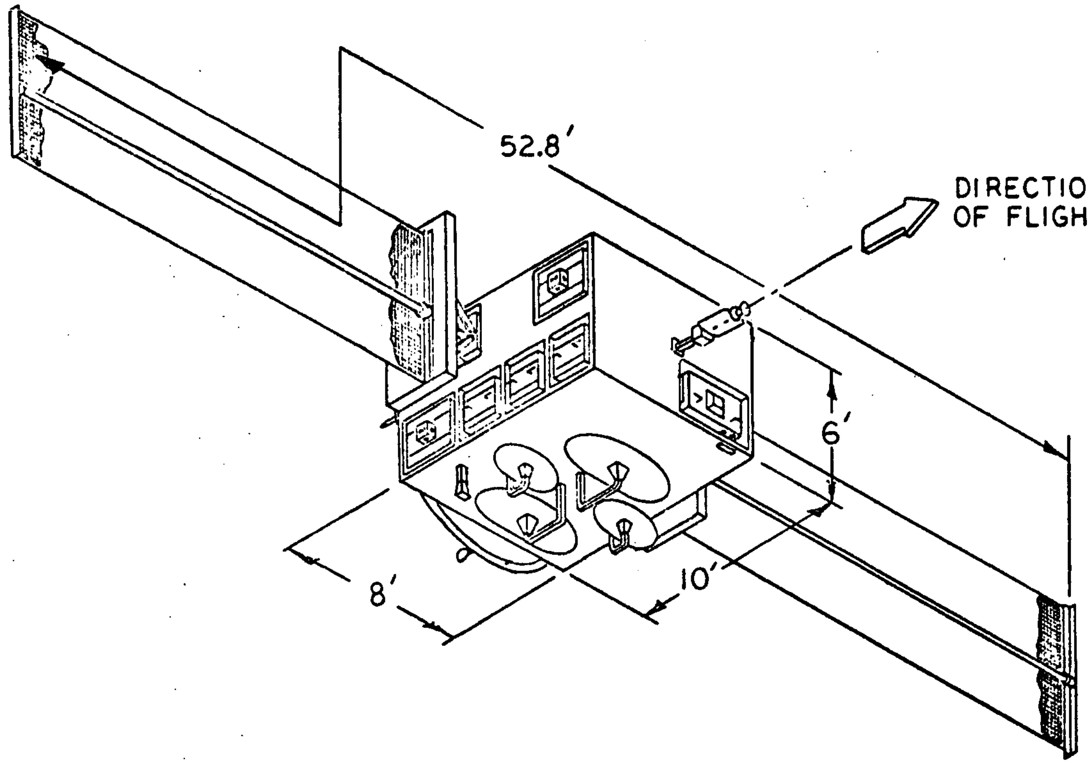


Fig. 7-12 General Configuration - Future Communications Satellite

<u>Spacecraft Subsystem Modules</u>

A-1 Attitude Control Module No. 1
A-3 Attitude Control Module No. 2
B-1 Battery Module No. 1
B-2 Battery Module No. 2
C-1 Solar Array Drive Module
D-1 Power Distribution Module
D-3 CDPI Module
E-2 S&C Sensing Module
E-4 Momentum Wheel Module
H-2 Attitude Control Module No. 3
H-4 Attitude Control Module No. 4

<u>Mission Equipment</u>

F-2 Transponder Module No. 1
F-4 Transponder Module No. 2
G-2 Transponder Module No. 3
G-4 Transponder Module No. 4

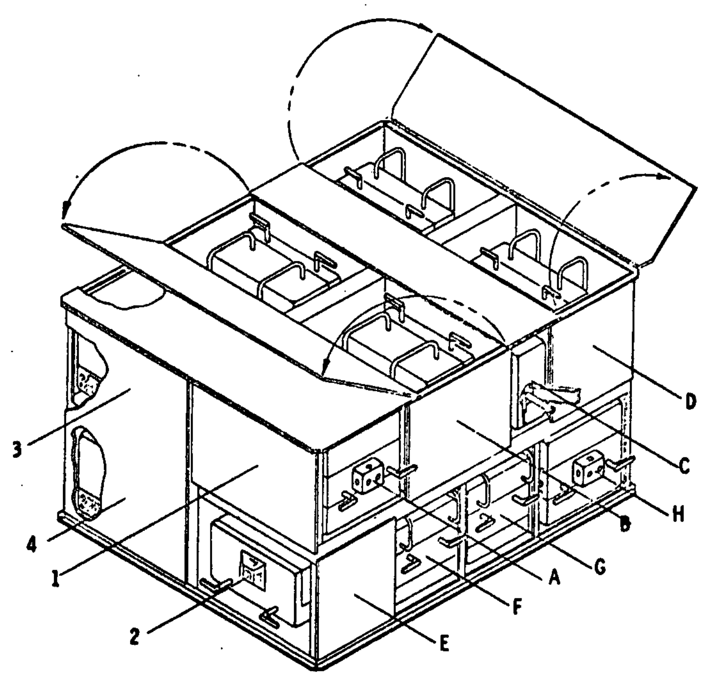


Fig. 7-13 Equipment Module Locations - Future Communication Satellite

outer-planet missions which have very special communications and electrical power requirements and very long mission life requirements.

7.1.7.2 Basic Modules and Variants. In establishing the standard module inventory, emphasis must be placed upon retaining the smallest possible quantity of basic modules, creating variants to these as necessary. A "variant" to a basic module is described as the addition, deletion, or replacement of a component (or components) within the basic module which will enhance the functional capability or design life of the module but will not alter its basic external functional interfaces nor its size (there may be a minor weight variation). The reason for restricting the quantity of basic modules is two-fold:

(1) The RDT&E cost, which must be amortized over the using spacecraft, is minimized with the least quantity of different modules requiring development (the variants add only a small increment of the cost required to develop another basic module)

(2) The logistics of supply, storage, refurbishment, and re-use are simplified with the smallest quantity of basic modules.

7.1.7.3 The Standard Module Inventory. Utilizing the following variables of data, a set of basic standard modules or variants thereof is developed for each subsystem of the mission-peculiar spacecraft (using the point-design modules as a reference):

a. The design life of the spacecraft (or MMD)

b. The special mission effects (such as spacecraft orientation to sun)

c. The functional requirement of the subsystem

A standard module inventory developed for the NASA Mission Model is briefly outlined in the following paragraphs for four subsystems.

7.1.7.4 S&C Subsystem Standard Modules. Figure 7-14 lists the S&C standard modules, comprising 6 basic and 3 variants (variants are indicated by the second dash number in the module identification number).

Module No.	Module Name	Equipment	Design Life (yrs.)	Wt. (lb)	Point Design Reference
S&C 1	Precision Sensing	Fixed Head Star Tracker (2) FHST Electronics (2) Three-Axis Rate Sensor Precision Equipment Mount Module Base & Cover Cables and Connectors	Basic: 1 Redundant (2 Modules): 3	105	IMSC PE-102 IMSC PE-106
S&C-2	Sensing	Sun Aspect Sensor (5) SAS Electronics Rate Gyro Package Control Electronics Module Base and Cover Cables and Connectors	5	64	IMSC PE-102 IMSC PE-106
S&C-2-1	Sensing	S&C-2 + High Altitude Horizon Sensor + HAHS Electronics	5	67	IMSC PE-102 IMSC PE-122
S&C-2-2	Sensing	S&C-2 + Low Altitude Horizon Sensor (2) + LAHS Electronics (2)	5	75	IMSC PE-102 IMSC PE-106
S&C-2-3	Sensing	S&C-2 + Velocity Control Accel. + VCA Electronics	3	70	IMSC PE-102 IMSC PE-106
S&C-3	Reaction Torque (Max angular momentum - 10 ft-lb-sec/wheel)	Reaction Wheel (3) Reaction Wheel Electronics Wheel Support & Safety Shield Magnetic Torquer (3) Magnetic Torquer Electronics Module Base and Cover Cables and Connectors	5	153	IMSC PE-102 IMSC PE-106
S&C-4	Momentum Wheel	Dual Gimbal Momentum Wheel Momentum Wheel Electronics Wheel Support & Safety Shield Module Base and Cover Cables and Connectors	7	228	IMSC PE-122 IMSC PE-126
S&C-5	Spin Axis Control	Earth Horizon Sensor (2) EHS Electronics Solar Aspect Sensor SAS Electronics Magnetic Torquer Electr. Module Base and Cover Cables and Connectors	1	53	IMSC PE-42 IMSC PE-47
S&C-6	Reaction Torque (Max. angular momentum = 50 ft-lb-sec/wheel)	Reaction Wheel (3) Reaction Wheel Electronics Wheel Support and Safety Shield Magnetic Torquer (3) Mag. Torquer Electronics Module Base and Cover Cables and Connectors	2	260	IMSC PE-146

Fig. 7-14 Standard Modules - Stabilization and Control

The equipment included in each module is listed as is the design life of the module and its weight. The point-design reference is also shown. The three variants to the basic S&C-2 Sensing module each involves the addition of sensors to the basic module to accommodate various modes of spatial sensing for different missions.

7.1.7.5 CDPI Subsystem Standard Modules. Figures 7-15a and 7-15b list the CDPI standard modules, comprising 6 basic modules and 5 variants. Variants CDPI-1-1, CDPI-2-1, CDPI-3-1, and CDPI-3-2 have been included to provide increased design life (internal redundancy within the module) of the module for longer-life spacecraft. Conversely, the variant CDPI-4-1 has been included to provide a down-rated version of CDPI-4 for shorter-life spacecraft.

7.1.7.6 EPS Subsystem Standard Modules. Figures 7-16a, 7-16b, and 7-16c list the EPS standard modules, comprising 7 basic modules and 17 variants. The large quantity of variants represent simple design extrapolations of basic solar array modules. The basic modules, EPS-1, EPS-2, and EPS-4 are extendable flexible solar arrays respectively 6 ft, 8 ft, and 10 ft wide. Each is designed to accept a flexible solar array of varying length. A single motorized extend/retract boom assembly accommodates the various lengths of extension required.

The variants represent the single hardware variable, a specified length of flexible solar array. These variants will allow a considerable saving in Unit cost of the modules; application of solar array square footage reasonably close to the mission requirement can be made, thereby saving about $2,000 to $3,000 per square foot cost of solar array overage (excess power above mission requirement). These solar array length variants account for 15 of the total 17 variants in the EPS standard module inventory.

7.1.7.7 ACS Subsystem Standard Modules. Figure 7-17 lists the ACS standard modules comprising 2 basic modules and one variant. ACS-1 is a cold gas (Freon) module. ACS-1-1 is a variant which provides a larger gas storage tank. ACS-2 is a hydrazine-propellant module, utilized for missions requiring a relatively large amount of total impulse.

Module No.	Module Name	Equipment	Design Life (Yrs.)	Wt. (lbs)	Point Design Reference
CDPI-1	K_u Band Communication	K_u Band TWTA (2) (50 watts output) K_u Band PLL Receiver K_u Band QPSK Modulator/Driver K_u Band Multicoupler Interface Unit (High Data Rate) Module Base and Cover Waveguide, Cables and Connectors	3	85	LMSC PE-103 LMSC PE-106
CDPI-1-1	K_u Band Communication	CDPI-I + K_u Band TWTA (2) + K_u Band PLL Receiver + K_u Band QPSK Modulator/Driver	5	120	LMSC PE-103 LMSC PE-106
CDPI-2	S-Band/VHF Communication	S-Band Transmitter (10 watts output) S-Band Receiver S-Band QPSK Modulator/Driver S-Band Multicoupler Delay Lock Loop Correlator(2) VHF Transmitter (5 watts output) VHF Receiver/Demodulator VHF Multicoupler Module Base and Cover Cables and Connectors	2	78	LMSC PE-103 LMSC PE-106
CDPI-2-1	S-Band/VHF Communication	CDPI-2 + VHF Transmitter (5 watts output) + VHF Receiver/ Demodulator	3	89	LMSC PE-103 LMSC PE-106
CDPI-3	Data Processing	Digital Computer (16K Memory) Interface Unit (Med and Low Data Rates) Timer Module Base and Cover Cables and Connectors	2	91	LMSC PE-103 LMSC PE-106
CDPI-3-1	Data Processing	CDPI-3 + Digital Computer (16K Memory)	3	98	LMSC PE-103 LMSC PE-106
CDPI-3-2	Data Processing	CDPI-3 - Digital Computer (16K Memory) + Hi-Rel Digital Computer, from CDPI-4 (Redundant 8K Memory)	5	91	LMSC PE-103 LMSC PE-123 (for computer) LMSC PE-106

Fig. 7-15a Standard Modules - CDPI Subsystem (1 of 2)

Module No.	Module Name	Equipment	Design Life (Years)	Wt. (lbs)	Point Design Reference
CDPI-4	Communication and Data Processing	Digital Computer, Hi-Rel (Redundant 8K Memory) Interface Unit Command Receiver (2) Decoder, Command Decoder, Failure Correct. Range Tone (Detect, Test and Switch)(2) Modulator and Summer (2) Transmitter, TM (1 watt)(2) Transmitter, Beacon (1w)(2) Hybrid Coupler (3) Misc. Amplifiers, Gates and Switches (5) Input Filter Module Base and Cover Cables and Connectors		67	IMSC PE-123 IMSC PE-126
CDPI-4-1	Communication and Data Processing	CDPI-4 -Command Receiver (1) -Range Tone (Detect, Test and Switch)(1) -Modulator and Summer (1) -Transmitter, TM (1) -Transmitter, Beacon (1) -Hybrid Coupler (3)	2	55	IMSC PE-123 IMSC PE-126
CDPI-5	Communication and Data Processing	S-Band TWTA S-Band Mod/Driver S-Band PLL Receiver S-Band Multicoupler S-Band Receiver Delay Lock Loop Correlator UHF Receiver/Demodulator Antenna Servo Electronics UHF Coupler Tape Transport Tape Transport Electronics Digital Computer Mission-Equip. Interface Unit Spacecraft Interface Unit Cables and Connectors Module Base & Covers	2	149	IMSC PE-133
CDPI-6	Antenna Module	Antenna (6 ft dish) Antenna Gimbal & Base Antenna Feed (S&K Bands) Rotary Joint (S-Band) Rotary Joint (K-Band) Antenna Servo Motors & Gears Antenna Servo Electronics Waveguide, Cables and Connectors	5	75	IMSC PE-103

Fig. 7-15b Standard Modules - CDPI Subsystem (2 of 2)

Module No.	Module Name	Equipment	Design Life (Yrs)	Weight (lbs)	Point Design Reference
EPS-1	Solar Power	Solar Array Container Extendable Boom Assy. (Max. Deployed Length=20 ft) Module Base and Cover Cables and Connectors Flexible Solar Array (6 ft wide x 4 ft long, 1800 2 x 4 cm cells)	5	102	LMSC PE-124 LMSC PE-126
EPS-1-1	Solar Power	EPS-1 -Flexible Solar Array (6 ft wide x 4 ft long) +Flexible Solar Array (6 ft wide x 6 ft long, 3000 2 x 4 cm cells)	5	108	LMSC PE-124 LMSC PE-126
EPS-1-2	Solar Power	EPS-1 -Flexible Solar Array (6 ft wide x 4 ft long) +Flexible Solar Array (6 ft wide x 7 ft long, 3600 2 x 4 cm cells)	5	111	LMSC PE-124 LMSC PE-126
EPS-1-3	Solar Power	EPS-1 -Flexible Solar Array (6 ft wide x 4 ft long) +Flexible Solar Array (6 ft wide x 8 ft long, 4200 2 x 4 cm cells)	5	114	LMSC PE-124 LMSC PE-126
EPS-1-4	Solar Power	EPS-1 -Flexible Solar Array (6 ft wide x 4 ft long) +Flexible Solar Array (6 ft wide x 10 ft long, 5400 2 x 4 cm cells)	5	120	LMSC PE-124 LMSC PE-126
EPS-1-5	Solar Power	EPS-1 -Flexible Solar Array (6 ft wide x 4 ft long) +Flexible Solar Array (6 ft wide x 11 ft long, 6000 2 x 4 cm cells)	5	123	LMSC PE-124 LMSC PE-126
EPS-1-6	Solar Power	EPS-1 -Flexible Solar Array (6 ft wide x 4 ft long) +Flexible Solar Array (6 ft wide x 13 ft long, 7200 2 x 4 cm cells)	5	129	LMSC PE-124 LMSC PE-126
EPS-1-7	Solar Power	EPS-1 -Flexible Solar Array (6 ft wide x 4 ft long) +Flexible Solar Array (6 ft wide x 18 ft long, 10200 2 x 4 cm cells)	5	144	LMSC PE-124 LMSC PE-126
EPS-2	Solar Power	Solar Array Container Extendable Boom Assy. (Max Deployed Length=35 ft) Module Base and Cover Cables and Connectors Flexible Solar Array (8 ft wide x 18 ft long, 13600 2 x 4 cm cells)	7	192	LMSC PE-124 LMSC PE-126

Fig. 7-16a Standard Modules - Electrical Power Subsystem (1 of 3)

Module No.	Module Name	Equipment	Design Life (Yrs)	Weight (lbs)	Point Design Reference
EPS-2-1	Solar Power	EPS-2 -Flexible Solar Array (8 ft wide x 18 ft long) +Flexible Solar Array (8 ft wide x 20 ft long, 15200 2 x 4 cm cells)	7	200	LMSC PE-124 LMSC PE-126
EPS-2-2	Solar Power	EPS-2 -Flexible Solar Array (8 ft wide x 18 ft long) +Flexible Solar Array (8 ft wide x 21 ft long, 16000 2 x 4 cm cells)	7	204	LMSC PE-124 LMSC PE-126
EPS-2-3	Solar Power	EPS-2 -Flexible Solar Array (8 ft wide x 18 ft long) +Flexible Solar Array (8 ft wide x 22 ft long, 16800 2 x 4 cm cells)	7	208	LMSC PE-124 LMSC PE-126
EPS-2-4	Solar Power	EPS-2 -Flexible Solar Array (8 ft wide x 18 ft long) +Flexible Solar Array (8 ft wide x 25 ft long, 19200 2 x 4 cm cells)	7	220	LMSC PE-124 LMSC PE-126
EPS-2-5	Solar Power	EPS-2 -Flexible Solar Array (8 ft wide x 18 ft long, +Flexible Solar Array (8 ft wide x 27 ft long, 20800 2 x 4 cm cells	7	228	LMSC PE-124 LMSC PE-126
EPS-2-6	Solar Power	EPS-2 -Flexible Solar Array (8 ft wide x 18 ft long) +Flexible Solar Array (8 ft wide x 33 ft long, 25600 2 x 4 cm cells)	7	252	LMSC PE-124 LMSC PE-126
EPS-3	Solar Power	Solar Panel (20.0 x 26.9 in., 396 2 x 4 cm cells)	1	6.4	LMSC PE-44

Fig. 7-16b Standard Modules - Electrical Power Subsystem (2 of 3)

Module No.	Module Name	Equipment	Design Life (yrs)	Weight (lbs)	Point Design Reference
EPS-4	Solar Power	• Solar Array Container • Extendable Boom Assy (Max Deployed Length = 55 ft) • Module Base & Cover • Cables & Connectors • Flexible Solar Array (10 ft wide x 32 ft long, 31000 2 x 4 cm cells)	7	320	IMSC PE-124 IMSC PE-126
EPS-4-1	Solar Power	EPS-4 - Flexible Solar Array (10 ft wide x 32 ft long) + Flexible Solar Array (10 ft wide x 40 ft long, 39000 2 x 4 cm cells)	7	360	IMSC PE-124 IMSC PE-126
EPS-4-2	Solar Power	EPS-4 - Flexible Solar Array (10 ft wide x 32 ft long) + Flexible Solar Array (10 ft wide x 53 ft long, 52000 2 x 4 cm cells)	7	425	IMSC PE-124 IMSC PE-126
EPS-5	Solar Array Drive	Drive Motor Assy Slip Ring Assembly Drive Electronics Module Base and Cover Cables and Connectors	5	76	IMSC PE-124 IMSC PE-126
EPS-6	Battery Power	NiCd Battery, 40AH (2) Charge Controller (2) Module Base and Cover Cables and Connectors	5 (Depth of discharge and Charge Discharge cycle to be controlled)	206	IMSC PE-104 IMSC PE-106 IMSC PE-124
EPS-6-1	Battery Power	EPS-6 -NiCd Battery, 40AH(2) +NiCd Battery, 20AH(2)	3	178	IMSC PE-104 IMSC PE-106 IMSC PE-124
EPS-7	Power Distribution	Power Distribution Unit Regulator Converter Module Base and Cover Cables and Connectors	5	120	IMSC PE-104 IMSC PE-106 IMSC PE-124
EPS-7-1	Power Distribution	EPS-7 + Regulator Converter	5	164	IMSC PE-104 IMSC PE-106 IMSC PE-124

Fig. 7-16c Standard Modules - Electrical Power Subsystem (3 of 3)

Module No.	Module Name	Equipment	Design Life (Yrs)	Weight (lbs)	Point Design Reference
ACS-1	Attitude Control - Cold Gas	Storage Tank, 16 in.OD Regulator Valve Assy Fill Valve Thruster Cluster Pressure Transducer (2) Temperature Sensor(2) ACS Electronics (part of S&C subsystem) Module Base and Cover Plumbing Cable and Connectors	3	101 (dry)	LMSC PE-105 LMSC PE-106
ACS-1-1	Attitude Control - Cold Gas	ACS-1 -Storage Tank, 16 in OD +Storage Tank, 22 in OD	3	136 (dry)	LMSC PE-105 LMSC PE-106
ACS-2	Attitude Control - Hydrazine	Storage Tank, 16 in OD Fill Valves (2) Thruster Cluster Filter Pressure Transducer Temperature Sensors (2) ACS Electronics (part of S&C Subsystem) Module Base and Cover Plumbing Cables and Connectors Heater, Strip	5	98 (dry)	LMSC PE-125 LMSC PE-126

Fig. 7-17 Standard Modules - Attitude Control Subsystem

7.2 Assignment of Standard Subsystem Modules to Mission-Peculiar Spacecraft

7.2.1 General Selection Approach

From the inventory of standard subsystem modules (from 7.1.7), a set of modules can be selected for each mission spacecraft in accordance with the following rules:

a. Select the least quantity of different modules required to accommodate each subsystem requirement.

b. Select a module, or multiple thereof with capability closest to the mission spacecraft requirement (at least equal but with minimum excess capability).

7.2.2 Specific Assignment of Standard Modules

The actual assignment of modules to a group of mission-peculiar spacecraft can be recorded on a listing of the various missions. Figure 7-18 shows a portion of such a listing. The module selected must match or exceed both the functional requirement and the design life of the mission-peculiar spacecraft. The module numbers are from the listing provided in previous par. 7.1.7. The complete listing of assignment of the standard modules to all spacecraft in the NASA Mission Model is included in an IMSC separate report "Parametric Analysis of Standard Space Hardware - Task 1.3 of Payload Effects Follow-On Study", IMSC-D154649, dtd 31 January 1972.

7.3 Weights and Sizes of Future Shuttle Payloads

Using the selected set of low-cost standard modules as a base, the weights and sizes of each of the future spacecraft can be estimated. Because the standard modules include all of the attributes of low-cost, refurbishable design; the weight of a future low-cost spacecraft, whether inclusive of standard modules or totally mission-peculiar, is the same.

7.3.1 Weight Estimating Methodology

a. Module Weight

The weight of each module is obtained from the module listing in 7.1.7. The module weight is a major factor in the total payload weight.

b. Spaceframe Weight

The weight of mission-peculiar spacecraft elements is estimated, using low-cost point designs as data points from which to extrapolate. Included are weights of the structural spaceframe and mechanisms, the integral electrical wiring, and the Environmental Control subsystem.

c. Total Dry Weight

This weight is the sum of the modules (exclusive of ACS propellant) and the spaceframe.

	Mission	S&C Mod. No.	S&C Qty.	CDPI Mod. No.	CDPI Qty.	EPS Mod. No.	EPS Qty.	ACS Mod. No.	ACS Qty.
1	Astronomy Explorer Des. Life = 3 yr LEO 200 W	S&C-1 S&C-2 S&C-3	2 1 1	CDPI-1 CDPI-2-1 CDPI-3-1 CDPI-6	1 1 1 1	EPS-1-1 EPS-5 EPS-6 EPS-7	2 2 1 1	ACS-1	4
2	Astronomy Explorer B Des. Life = 3 yr HE 200 W	S&C-2-1 S&C-4	1 1	CDPI-2 CDPI-3-1 CDPI-6	1 1 1	EPS-1 EPS-5 EPS-6 EPS-7	2 2 1 1	ACS-1	4
3	Magnetosphere-Low Des. Life = 1 yr LEO 200 W	S&C-5	1	CDPI-2 CDPI-3	1 1	EPS-3 EPS-6 EPS-7	32 1 1	NA	
4	Magnetosphere-Middle Des. Life = 1 yr HE 100 W	S&C-5	1	CDPI-2 CDPI-3	1 1	EPS-3 EPS-6-1 EPS-7	8 1 1	NA	
5	Magnetosphere-Upper Des. Life = 1 yr HE 100 W	S&C-5	1	CDPI-2 CDPI-3	1 1	EPS-3 EPS-6-1 EPS-7	8 1 1	NA	
6	OSO Des. Life LEO 500 W	S&C-1 S&C-2 S&C-3	1 1 1	CDPI-1 CDPI-2 CDPI-3 CDPI-6	1 1 1 1	EPS-1-5 EPS-6 EPS-7	2 2 1	ACS-1	4
75	TIROS Op. Met. Des Life = 4 yr LEO 300 W	S&C-2-2 S&C-4	1 1	CDPI-2-1 CDPI-3-2 CDPI-6	2 1 1	EPS-1-5 EPS-6 EPS-7-1	2 2 1	ACS-1	4
76	Sync. Met. Des. Life = 2 yr Syneq 400 W	S&C-1 S&C-2 S&C-3	2 1 1	CDPI-2 CDPI-3	1 1	EPS-1-2 EPS-5 EPS-6-1 EPS-7	2 2 1 1	ACS-2	4
77	Polar ER Des. Life = 2 yr LEO 1000 W	S&C-1 S&C-2 S&C-3	2 1 1	CDPI-1 CDPI-2 CDPI-3 CDPI-6	1 1 1 1	EPS-2-5 EPS-6 EPS-7	2 3 1	ACS-1-1	4
78	Sync. ER Des. Life = 3 yr Syneq 400 W	S&C-1 S&C-2 S&C-3	2 1 1	CDPI-1 CDPI-4-1	1 1	EPS-1-2 EPS-5 EPS-6 EPS-7	2 2 1 1	ACS-2	4

Fig. 7-18 Assignment of Standard Modules to Mission-Peculiar Spacecraft (Partial Listing)

d. Expendables

The approximate total impulse for performing the particular mission is estimated. This is converted to propellant weight, allowing reasonable overages.

e. Experiment/Sensor Weight

The baseline (NASA-supplied) weight of the experiment/sensor package is extrapolated, assuming a low-cost design has been implemented (increased weight and volume; reduced cost). Specific analysis should be done to isolate those hardware elements which can show a cost benefit from weight increase.

f. Weight Summary

The aforementioned weights can be summarized on a chart for each mission. Figure 7-19 is an example of a summary. The numbers on the left margin identify the mission.

Future Payload Weights

	Total Module Weight	Spaceframe, ECS, etc. Weight	Total Dry Weight	Expendables	Total S/C Weight	Exper. Weight	Payload Weight
1	1900	2090	3990	140	4130	425	4560
2	1665	1080	2745	140	2890	425	3320
3	781	860	1640	-	1640	340	1980
4	571	370	940	-	940	310	1250
5	571	370	940	-	940	260	1200

Fig. 7-19 Weight Summary of Future Payloads

7.3.2 Volume Estimates

A significant result of application of standard modules to many different missions has been the emergence of what appears to be also a potential standardization of spacecraft structures and attendant payload overall dimensions. With the exception of the large astronomical observatories and a few very specialized mission configurations, the mission payloads can be "standardized" into rectangular arrangements, approximately 8 ft wide and 6 ft high with length varying from 7 ft to 20 ft.

The estimates for the dimensions of the future payloads are influenced principally by two factors:

(1) The cumulative volume of the spacecraft equipment modules, allowing extra volume for later changes to module-set complement or addition of modules for expanded or updated experiment support.

(2) The probable volume of a low-cost, modularized experiment package.

7.3.3 Future Payload Weights and Dimensions

Figure 7-20 is a listing of estimated weights and dimensions of future payloads for missions on the NASA Mission Model.

7.4 Design of Standard Spacecraft

7.4.1 Basic Types of Standard Spacecraft

Standard Spacecraft are multi-mission spacecraft; they are able to support several different mission experiment packages, one package on any particular mission. To do this the Standard Spacecraft may either (1) carry the standard subsystem modules satisfying the most demanding mission every time, or (2) may make use of alternate standard modules for each mission wherein the module capability more closely matches the requirements of the particular mission to be flown.

Among the various possibilities for low-earth-orbit (LEO) Standard Spacecraft there are two "natural" contenders: (a) One type for low inclination orbits to support the observatory missions LST, LRO, LSO, and HEAO, and (b) another type for sun-synchronous orbits to support the EOS-type missions. The availability of both basic modules and variants allows the subsystem over-capability to be at a minimum. This approach to subsystem implementation is so flexible, that the only basic differences between Standard Spacecraft types will be in the physical experiment package support, which will be reflected mainly in the detail spacecraft structural design.

	Mission	Des. Life (yr)	NASA Baseline Payload (lb)	Future Payload Weight (2) (lb)			Future Payload Size (ft) (1) (2)
				Exper.	S/C	Payload Total	
1	Astronomy Explorer	3	860	430	4130	4560	8W x 6H x 17L
2	Astronomy Expl. B	3	860	430	2890	3320	10W x 6H x 12L
3	Magnetosphere-Low	1	1160	340	1640	1980	6 Dia x 8L
4	Magnetosphere-Mid.	1	965	310	940	1250	5 Dia x 6L
5	Magnetosphere-Upper	1	580	260	940	1200	5 Dia x 6L
6	OSO	1	1900	970	4110	5080	10W x 6H x 17L
7	Gravity Relativity	1	1450	680	4400	5080	10W x 6H x 17L
8	Gravity Relativity	1	485	400	2820	3220	10W x 6H x 12L
9	Radio Interferometer	3	10350	6630	3140	9770	14 Dia x 12L
10	Solar Orbit Pair-A	5	1820	1280	2990	4270	10W x 6H x 14L
11	Solar Orbit Pair-B	5	2440	1280	3600	4880	10W x 6H x 14L
12	Optical Interferom.	5	3040	1280	3220	4500	10W x 6H x 14L
13	HEAO	2	20750	18000	8090	26090	14 Dia x 40L
15	LST	2	21300	14000	8260	22260	14 Dia x 45L
17	LSO	2	26810	12000	7940	19940	14 Dia x 55L
19	LRO	2	19300	13000	8880	21880	14 Dia x 45L
21	Polar EOS	2	2500	1600	5820	7420	10W x 6H x 22L
22	SEO	2	1000	600	3490	4090	10W x 8H x 8L
23	Earth Physics	2	580	400	3660	4060	10W x 6H x 15L
24	Sync. Met.	2	1000	450	3560	4010	10W x 8H x 8L
25	TIROS	5	1000	600	4190	4790	10W x 6H x 17L
26	Polar ERS	2	2500	1600	5820	7420	10W x 6H x 22L
27	Sync. ER	2	1000	520	3490	4010	10W x 8H x 8L
28	ATS	5	7950	3000	4000	7000	10W x 8H x 14L
29	Small ATS-B	1	600	600	3370	3970	10W x 8H x 8L
30	Small ATS-A	1	600	600	4020	4620	10W x 6H x 17L
31	Coop ATS-A	2	820	500	3540	4040	10W x 8H x 8L
32	Coop ATS-B	2	820	500	4240	4740	10W x 6H x 17L
33	Medical Network	5	2000	1000	3720	4720	10W x 8H x 10L
34	Educ. Broadcast	5	3400	1700	4380	6080	10W x 6H x 12L
35	F.O. Sys. Demonst.	5	2000	1000	3640	4640	10W x 8H x 10L
36	TDRS	3	2300	1200	3230	4430	10W x 8H x 10L
50	Mars Viking	1	7570	4200	21800	26000	14 Dia x 20L
51	Mars Sample Ret.	3	10290			NA	NA
52	Venus Expl.-Orb.	1	970	600	9380	9980	8W x 6H x 20L
53	Venus Radar Map.	2	7630	800	10980	11780	8W x 8H x 20L
54	Venus Explor.-Ldr.	1	7260	2400	15970	18370	14 Dia x 12L
55	Jup. Pioneer Orb.	2	900			NA	NA
56	Grand Tour (JUN)	9	1480			NA	NA
57	Jup. TOPS Orb.	3	3180			NA	NA
58	Uranus TOPS Orb.	7	3580			NA	NA
59	Asteroid Survey	4	1840			NA	NA
60	Comet Rendezvous	4	1200			NA	NA
70	COMSAT	5	1420	600	3210	3810	10W x 8H x 8L
71	US Dom. Comm.	7	3425	1700	4070	5770	10W x 8H x 12L
72	Foreign Dom.Comm.	5	1000	500	2900	3400	10W x 8H x 7L
73	NAV/Traf.Contr. B	5	700	500	2730	3230	10W x 8H x 7L
74	NAV/Traf.Contr. A	5	700	500	2600	3100	10W x 8H x 7L
75	TIROS Op. Met.	4	1000	600	4190	4790	10W x 6H x 17L
76	Sync. Met.	2	1000	450	3440	3890	10W x 8H x 8L
77	Polar ER	2	2500	1600	5820	7420	10W x 6H x 22L
78	Sync. ER	3	1000	520	3490	4010	10W x 8H x 8L

(1) Size inclusive of spacecraft plus experiment packages
(2) Future payload incorporates low-cost and refurbishable features

Fig. 7-20 Future Payload Weights & Dimensions

Assuming the implementation of the aforementioned two types of Standard Spacecraft, there remains a group of LEO missions with small experiment packages, and with experiment-support requirements that could be satisfied by a third standard spacecraft type.

Herein, only LEO missions are considered for the application of Standard Spacecraft. Future applications, however, may be expanded readily to cover synchronous orbit and other high energy orbit missions. For example, the LMSC-designed Communications Satellite (Ref. PE-126 - see Volume II) may be adapted with minor changes to handle a variety of communications missions in Syneq orbit.

7.4.2 Typical Standard Spacecraft Designs

Preliminary point designs of three Standard Spacecraft have been documented in LMSC Engineering Memos (copies are in Volume II of the Design Guide):

Standard Spacecraft	Engineering Memo
Standard Earth Observatory Spacecraft	PE-156
Standard Astronomical Observatory Spacecraft	PE-146
Standard Communication Spacecraft	PE-126

The following paragraphs provide the principal features of these spacecraft.

7.4.2.1 Standard Earth Observatory Spacecraft. A single basic Standard Spacecraft, numbered SSC-3, can support three NASA missions; a variant of this spacecraft, SSC-3-1, can support an additional two missions:

Standard Spacecraft	NASA Mission	Total Flights
SSC-3	21 Polar EOS 26 Polar ERS 77 Polar ERS	40
SSC-3-1	25 TIROS 75 TOS Met	15

The single spacecraft can therefore support 5 missions and 55 flights. The potential for RDT&E cost saving is quite significant.

The basic spacecraft is configured to support earth resources missions; the variant will support meteorological missions. The description of the typical EOS presented in par. 7.1.6 is pertinent also to the SSC-3 Standard Spacecraft with the following minor modifications:

a. A second Precision Sensing module, S&C-1, is added to the Stabilization and Control Subsystem to increase the ultimate accuracy of the subsystem.

b. The rigid-panel solar array installation is replaced by two flexible solar array modules, EPS-2-5. Modularization of the solar array facilitates the substitution of other flexible solar array modules such as EPS-2-6 or EPS-1-5 as the total average power requirement of the spacecraft varies from mission to mission.

c. The four Attitude Control modules, type ACS-1 are replaced by four ACS-1-1 modules, each of which has a 22" diameter gas storage tank rather than the 16" diameter storage tank of the ACS-1 module. The quantity of cold gas propellant (Freon 14) per module is increased from 35 lbs to 100 lbs to provide for drag makeup in a 270 nm circular orbit. (The point-design Earth Observatory Satellite described by PE-126 (Volume II) was designed initially to operate in a 485 nm circular orbit with attendant lower drag).

7.4.2.2 Standard Astronomical Observatory. A single basic Standard Spacecraft, identified SSC-2, with two minor variants for the electrical power subsystem; can accommodate all of the large-observatory missions:

Standard Spacecraft	NASA Mission
SSC-2	13 HEAO-C
	15 Large Stellar Telescope
SSC-2-1	17 Large Solar Observatory
SSC-2-2	19 Large Radio Observatory

A preliminary conceptual design of a Typical Standard Astronomical Observatory Spacecraft is described in LMSC Engineering Memo PE-146 (Volume II). The following paragraphs provide an outline of its principal characteristics.

a. General Description

The Standard Astronomical Observatory Satellite which includes spacecraft and experiment package, is expected to be flown in a standard circular orbit at 324 nm altitude and 30° inclination to facilitate the revisit of several observatories by a single Shuttle flight.

Figure 7-21 depicts the general configuration of a typical Standard Large Astronomical Observatory Satellite with an LST payload shown in phantom.

b. Solar Array Orientation

In flight the +X axis is pointed at a target star and the payload is rolled about the X axis until the center line of the solar array is normal to the satellite/sun line. The solar array is then rotated about its center line until the surface of the array is normal to the solar radiation. For every pointing direction of the LST an appropriate combination of satellite roll and solar array rotation maintain the array surface normal to the solar radiation.

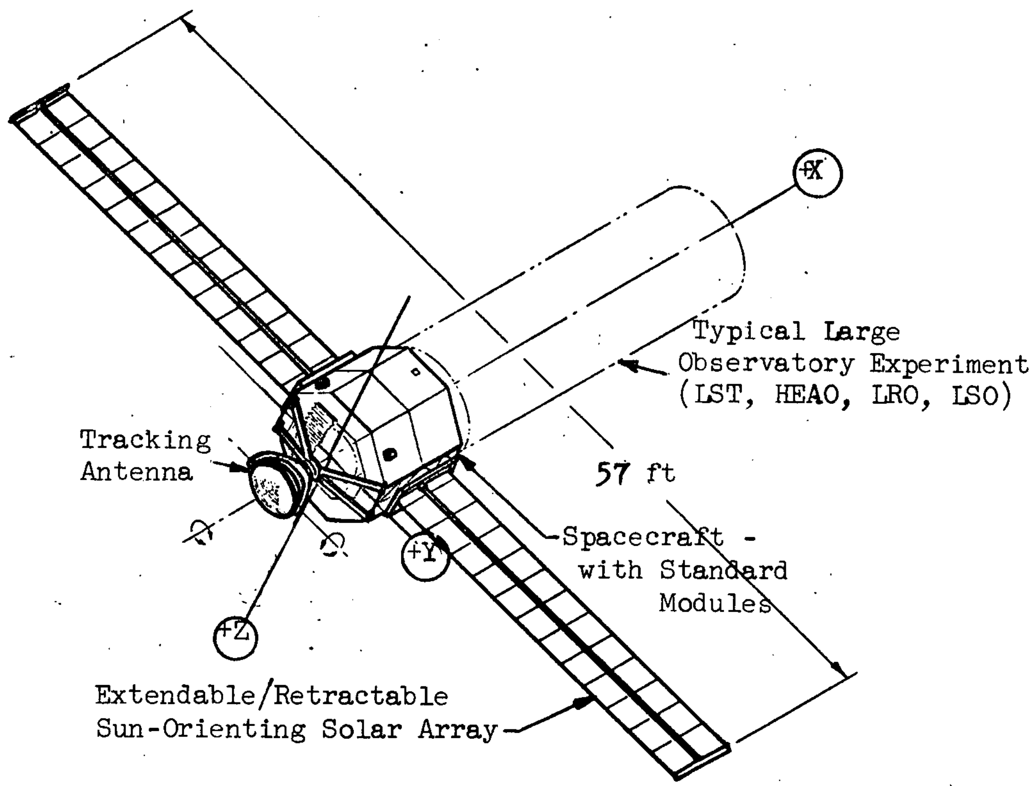


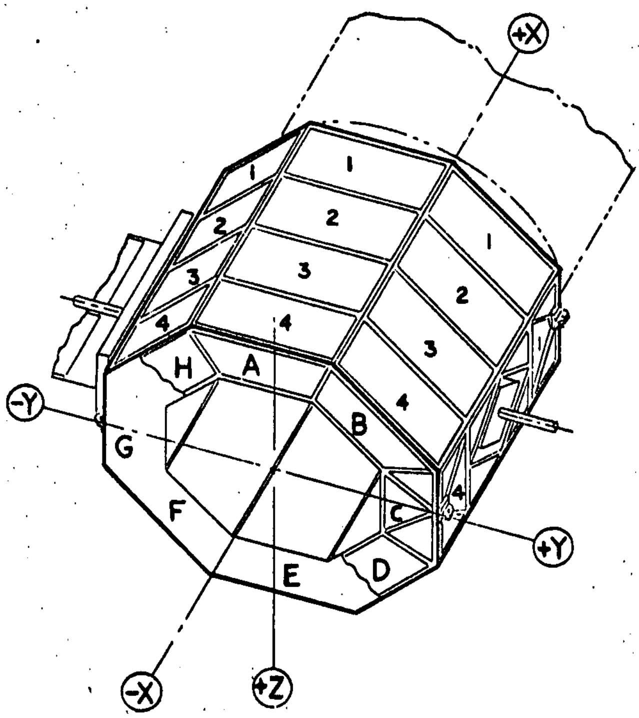
Fig. 7-21 Typical Standard Spacecraft for Large Astronomical Observatory

c. Communication Link to TDRS

The gimballed antenna provides communication with the Tracking and Data Relay Satellite (TDRS) network. Rotation of the antenna about the X axis and about an axis parallel to the Y axis ensure access to one TDRS for every pointing direction of the LST, thus providing real-time communication and eliminating any requirement for bulk data storage on-board the observatory spacecraft. The antenna mount is hinged to permit access to the large central payload compartment.

d. Spacecraft Structure

The spacecraft structure and a universal adapter to fit the four different experiment packages, designed in accordance with the low-cost design guidelines in Section 6.2, is made up of commercially-available aluminum sheet and extrusions. Twenty-four compartments for standard subsystem modules are provided as shown in Fig. 7-22.



Location	Module	Location	Module
A-1	Secondary Sensing	E-1	Empty
A-2	Empty	E-2	Empty
A-3	S-Band/VHF Communication	E-3	Power Distribution
A-4	K_u Band Communication	E-4	Battery Power
B-1	Empty	F-1	Precision Sensing
B-2	Empty	F-2	Battery Power
B-3	Reaction Torque	F-3	Battery Power
B-4	Attitude Control	F-4	Attitude Control
C-1	Empty	G-1	Empty
C-2	Solar Array Drive	G-2	Solar Array Drive
C-3	Empty	G-3	Empty
C-4	Empty	G-4	Empty
D-1	Precision Sensing	H-1	Empty
D-2	Battery Power	H-2	Data Processing
D-3	Battery Power	H-3	Reaction Torque
D-4	Attitude Control	H-4	Attitude Control

Fig. 7-22 Subsystem Module Locations - Standard Large Astronomical Spacecraft

The equipment compartments are closed by non-structural thermal-protection doors.

Four machined fittings are provided for handling the spacecraft and for supporting it in the Shuttle cargo bay. The experiment package for HEAO, LST, LSO, or LRO is assumed to be designed to mate with an interchangeable bolt circle on the adapter ring. The internal cavity of the spacecraft structure provides for major additional experiment modules such as the LST on-axis instrument modules, which may be serviced by opening a large accordion-pleat type door on the aft end of the spacecraft.

7.4.2.3 Standard Spacecraft for Miscellaneous LEO Missions. A third basic Standard Spacecraft, designated SSC-1, with five variants, SSC-1-1 through SSC-1-5 can support six missions:

Standard Spacecraft	NASA Mission	Total Flights
SSC-1	6 OSO	1
SSC-1-1	1 Astronomy Explorer A	15
SSC-1-2	7 Gravity/Relativity	2
SSC-1-3	23 Earth Physics	7
SSC-1-4	30 Small ATS	12
SSC-1-5	32 Coop ATS	2

A common set of Stabilization & Control modules is used in all versions. The Sensing Module S&C-1 is redundified for missions 1, 23, and 32. All variants are created by alternate sets of CDPI and EPS modules.

This spacecraft represents a single RDT&E cost to accommodate 6 different missions and, if all missions were accomplished with expendable spacecraft; represents a total of 39 spacecraft.

7.4.2.4 Standard Communication Satellite. A point design of a typical Standard Spacecraft for synchronous equatorial orbit is represented by the long-life (5 years or more) Communications Satellite discussed in par. 7.1.6. This preliminary design and analysis leads to the conclusion that a Standard Spacecraft, incorporating standard subsystems, can be readily designed to perform all of the Synchronous orbit earth observation missions.

The basic development of such a Syneq orbit Standard Spacecraft would parallel the approaches described for the LEO Standard Spacecraft.

7.4.3 Application of Standard Subsystems to Standard Spacecraft

The three basic Standard Spacecraft for LEO missions described in par. 7.4.2.1, 7.4.2.2, and 7.4.2.3 were configured to accept standard modules.

Standard modules have been selected for each of the four subsystems and for each of the missions performed by the three basic Standard Spacecraft. Figures 7-23a and 7-23b are a tabulation listing the assigned standard modules (previously described in detail in par. 7.1.7).

7.5 Design of Cluster Spacecraft

7.5.1 Common Orbits for Spacecraft

In the general approach to standardization of space hardware, it is first desirable to inspect the set of missions to determine if some consolidation of mission orbits is feasible. This was done for a group of LEO missions listed in the NASA Mission Model. These missions are listed in four groups on Fig. 7-24.

A large group of missions can be combined into a 30° low-inclination orbit at 600 km (Combination A). Other missions can be grouped into two different sun-synchronous orbits at 97.4° inclination and 500 km; one for noon-midnight and the other for dawn/dusk (combinations B and C, respectively). The 5 remaining missions are not readily combinable. This mission combination lays the base for application of Cluster Spacecraft.

Std. S/C No.	Mission No.	Mission Name	S&C Subsystem Module No.	S&C Qty.	CDPI Subsystem Module No.	CDPI Qty.	EPS Subsystem Module No.	EPS Qty.	ACS Subsystem Module No.	ACS Qty.
SSC-1 (Basic)	6	OSO	S&C-1 S&C-2 S&C-3	1 1 1	CDPI-1 CDPI-2 CDPI-3 CDPI-6	1 1 1 1	EPS-1-5 EPS-6 EPS-7	2 2 1	ACS-1	4
SSC-1-1	1	Astronomy-Explorer A	S&C-1 S&C-2 S&C-3	2 1 1	CDPI-1 CDPI-2 CDPI-2-1 CDPI-3 CDPI-3-1 CDPI-6	1 0 1 0 1 1	EPS-1-1 EPS-5 EPS-6 EPS-7	2 2 1 1	ACS-1	4
SSC-1-2	7	Gravity/ Relativity	S&C-1 S&C-2 S&C-3	1 1 1	CDPI-1 CDPI-2 CDPI-3 CDPI-6	0 1 1 0	EPS-1-5 EPS-2-2 EPS-6 EPS-7	0 2 3 1	ACS-1	4
SSC-1-3	23	Earth Physics	S&C-1 S&C-2 S&C-3	2 1 1	CDPI-1 CDPI-2 CDPI-3 CDPI-6	0 1 1 1	EPS-1-5 EPS-6 EPS-7	2 1 1	ACS-1	4
SSC-1-4	30	Small ATS	S&C-1 S&C-2 S&C-3	1 1 1	CDPI-1 CDPI-2 CDPI-3 CDPI-6	0 1 1 1	EPS-1-5 EPS-1-7 EPS-6 EPS-7	0 2 2 1	ACS-1	4
SSC-1-5	32	Coop ATS	S&C-1 S&C-2 S&C-3	2 1 1	CDPI-1 CDPI-2 CDPI-3 CDPI-6	0 1 1 1	EPS-1-5 EPS-1-7 EPS-6 EPS-7	0 2 2 1	ACS-1	4
SSC-2 (Basic)	13	HEAO-C	S&C-1 S&C-2 S&C-6	2 1 2	CDPI-1 CDPI-2 CDPI-3 CDPI-6	1 1 1 1	EPS-2-4 EPS-5 EPS-6 EPS-7	2 2 5 1	ACS-1-1	4
SSC-2 (Basic)	15	Large Stellar Telescope	S&C-1 S&C-2 S&C-6	2 1 2	CDPI-1 CDPI-2 CDPI-3 CDPI-6	1 1 1 1	EPS-2-4 EPS-5 EPS-6 EPS-7	2 2 5 1	ACS-1-1	4
SSC-2-1	17	Large Solar Observa-tory	S&C-1 S&C-2 S&C-6	2 1 2	CDPI-1 CDPI-2 CDPI-3 CDPI-6	1 1 1 1	EPS-2-4 EPS-5 EPS-6 EPS-7	2 0 5 1	ACS-1-1	4
SSC-2-2	19	Large Radio Observa-tory	S&C-1 S&C-2 S&C-6	2 1 2	CDPI-1 CDPI-2 CDPI-3 CDPI-6	1 1 1 1	EPS-2-4 EPS-2-6 EPS-5 EPS-6 EPS-7	0 2 2 6 1	ACS-1-1	4

Fig. 7-23a Standard Modules Assigned to LEO Standard Spacecraft (Sheet 1 of 2)

Std. S/C No.	Mission No.	Mission Name	S&C Subsystem Module No.	Qty.	CDPI Subsystem Module No.	Qty.	EPS Subsystem Module No.	Qty.	ACS Subsystem Module No.	Qty.
SSC-3 (Basic)	26	Polar ERS	S&C-1 S&C-2 S&C-3	2 1 1	CDPI-1 CDPI-2 CDPI-3 CDPI-6	2 1 1 1	EPS-2-5 EPS-6 EPS-7	2 3 1	ACS-1-1	4
SSC-3 (Basic)	77	Polar ERS	S&C-1 S&C-2 S&C-3	2 1 1	CDPI-1 CDPI-2 CDPI-3 CDPI-6	2 1 1 1	EPS-2-5 EPS-6 EPS-7	2 3 1	ACS-1-1	4
SSC-3 (Basic)	21	Polar EOS	S&C-1 S&C-2 S&C-3	2 1 1	CDPI-1 CDPI-2 CDPI-3 CDPI-6	2 1 1 1	EPS-2-5 EPS-6 EPS-7	2 3 1	ACS-1-1	4
SSC-3-1	25	TIROS	S&C-1 S&C-2 S&C-2-2 S&C-3 S&C-4	0 0 1 0 1	CDPI-1 CDPI-2 CDPI-2-1 CDPI-3 CDPI-3-2 CDPI-6	0 0 2 0 1 1	EPS-1-5 EPS-2-5 EPS-6 EPS-7 EPS-7-1	2 0 2 0 1	ACS-1 ACS-1-1	4 0
SSC-3-1	75	TOS Met	S&C-1 S&C-2 S&C-2-2 S&C-3 S&C-4	0 0 1 0 1	CDPI-1 CDPI-2 CDPI-2-1 CDPI-3 CDPI-3-2 CDPI-6	0 0 2 0 1 1	EPS-1-5 EPS-2-5 EPS-6 EPS-7 EPS-7-1	2 0 2 0 1	ACS-1 ACS-1-1	4 0

Note: Modules shown as [XX-X] are variants to the basic SSC complement of standard subsystem modules.

Fig. 7-23b Standard Modules Assigned to LEO Standard Spacecraft

Comb.	Composite Mission Alt. (KM)	Composite Mission Incl.	Composite Mission Orbit Time	Baseline Missions Fleming No.	Mission Name	Alt. (KM)	Incl.
A	600	30°	-	1	Astronomy Explorer A	500	28.5°
				6	OSO	650	30°
				13	HEAO-C	650	28.5°
				15	LST	650	28.5°
				17	LSO	650	30°
				19	LRO	650	30°
B	500	97.4° SS	Noon/ Mid-night	21	Polar EOS	930	99° SS
				25	TIROS	1300	101° SS
				26	Polar ERS	930	99° SS
				75	TOS Met.	1300	101° SS
				77	Polar ERS	930	99° SS
C	500	97.4° SS	Dawn/ Dusk	26	Polar ERS	930	99° SS
				77	Polar ERS	930	99° SS
-	Not Combinable			3	Magnetosphere - Low	3520/ 260	28.5° - 90°
				7	Gravity/Relativity	930	90°
				23	Earth Physics	740	90°
				30	Small ATS	5550/ 555	90°
				32	Co-op ATS		

Fig. 7-24 Candidates for LEO Mission Combination

7.5.2 Concept of Cluster Spacecraft

Cluster Spacecraft are multimission spacecraft that are capable of supporting several missions at the same time. Within limits, their modular subsystem complements can be changed in orbit to match the requirements of later (as yet undefined) missions.

The prerequisites for the initiation of a Cluster Spacecraft concept are the coincidence of mission orbits, and of flight schedules for compatible payloads. The cost benefits of Cluster Spacecraft, which are in addition to the potential savings due to equipment module standardization, are derived from the sharing of Shuttle transportation costs for the placement and servicing of multiple payloads. With the sharing of transportation costs among missions, the cost of in-orbit payload repair and refurbishment also is reduced.

The special advantages of the Cluster Spacecraft lie in the facts that (1) no maneuvering is required by the Shuttle to visit a number of experiments simultaneously, (2) that the number of spacecraft placement launches are reduced, and (3) that the total subsystem support requirements per Cluster mission are smaller than the sum of those for the equivalent mission-peculiar spacecraft.

7.5.3 Types of Cluster Spacecraft

One set of missions seems to be predestined to fly on a Cluster; it includes all the LEO earth observation, earth resources, and meteorology missions. The sensors for these missions operate with a minimum of mutual interference.

Another group of missions with similar support requirements are the large astronomical observatories in low-inclination orbit.

It has been determined that it is feasible to design a Cluster Spacecraft to perform the HEAO and LST missions simultaneously and have additional capability to support Astronomy Explorer or OSO missions. This Cluster Spacecraft has been designated CSC-1.

The same basic Cluster Spacecraft design will also perform the LSO and LRO missions simultaneously with minor exchange of electrical modules to provide additional power. This Cluster variant is designated CSC-1-1.

7.5.4 Application of Standard Modules to Cluster Spacecraft

Figure 7-25 identifies two basic Cluster Spacecraft CSC-1 and CSC-2, and the one variant, namely CSC-1-1. The standard modules assigned to each are identified (standard modules described in 7.1.7).

Cluster S/C No.	Mission No.	Mission Name	S&C Subsystem Module No.	S&C Qty.	CDPI Subsystem Module No.	CDPI Qty.	EPS Subsystem Module No.	EPS Qty.	ACS Subsystem Module No.	ACS Qty.
CSC-1 (Basic)	1 6 13 15	Stellar Oriented Observatory Cluster	S&C-1 S&C-2 S&C-6	2 1 3	CDPI-1 CDPI-2 CDPI-3 CDPI-6	1 1 1 1	EPS-4* EPS-5 EPS-6 EPS-7	2 2 8 1	ACS-1-1	4
CSC-1-1	17 19	Solar/ Stellar Oriented Observ. Cluster	S&C-1 S&C-2 S&C-6	2 1 3	CDPI-1 CDPI-2 CDPI-3 CDPI-6	1 1 1 1	EPS-4* EPS-4-1* EPS-5 EPS-6 EPS-7	0 2 0 10 1	ACS-1-1	4
CSC-2 (Basic)	21 25 26 77	Sun-Sync. Earth Obs. Cluster	S&C-1 S&C-2 S&C-6	2 1 1	CDPI-1 CDPI-2 CDPI-3 CDPI-6	2 1 1 1	EPS-4-2* EPS-6 EPS-7	2 8 1	ACS-1-1	4

Notes: * The three different modules marked with asterisk are new module variants required only for the cluster spacecraft.

** Modules shown as $\boxed{\text{XX-X}}$ are variants to the basic SSC complement.

Fig. 7-25 LEO Cluster Spacecraft & Standard Module Assignment

7.5.5 Typical Cluster Spacecraft Designs

Preliminary conceptual designs of typical Cluster Spacecraft have been documented in LMSC Engineering Memos (see Volume II):

Cluster Spacecraft	Engineering Memo
Cluster Astronomical Observatory Spacecraft	PE-186
Cluster Earth Observatory Spacecraft	PE-166

Principal characteristics of these spacecraft are outlined in the following paragraphs.

7.5.5.1 Cluster Astronomical Observatory Spacecraft.

a. General Configuration

The general configuration of a typical Cluster Astronomical Observatory (CAOS) is shown in Fig. 7-26.

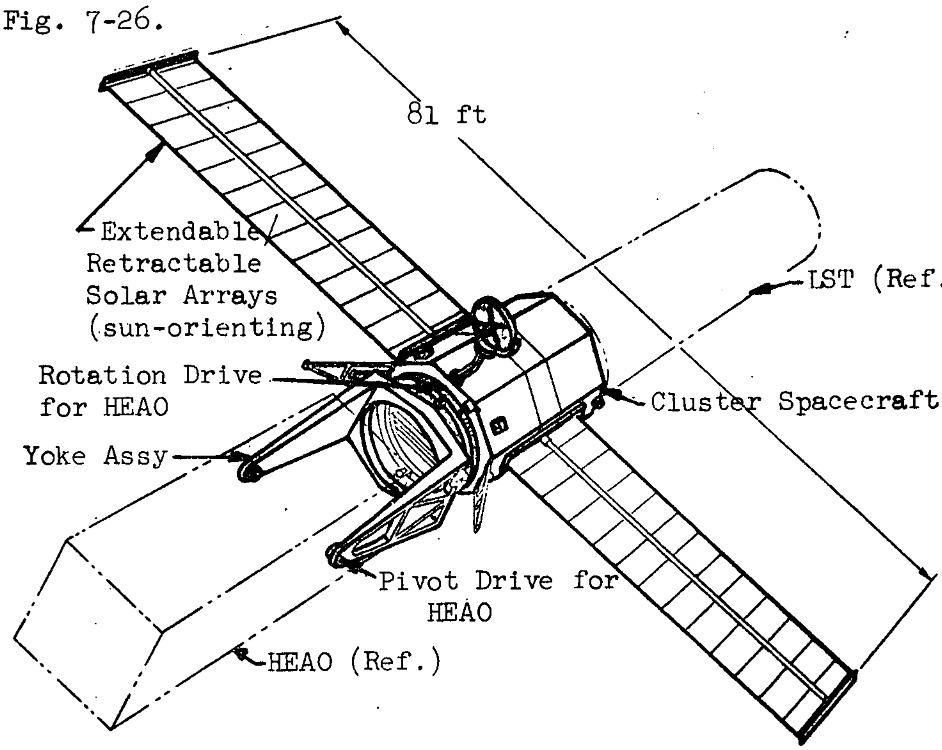


Fig. 7-26 Cluster Astronomical Observatory

The basic spacecraft is a derivation from the Standard Astronomical Observatory Spacecraft (SAOS) described in par. 7.4.2.2. The length of the spacecraft structure is increased by 2.5 ft to provide six more compartments for subsystem modules or for auxiliary experiment sensors.

b. Tracking Solar Array and Antennas

The solar array is larger than that of the SAOS to generate the higher average power required. It tracks the sun similarly. Two tracking antennas are installed rather than one, one on the sunlit side and one on the side opposite, to provide continuous communication with the TDRS network regardless of satellite attitude. The switching of signals from one antenna to the other is under control of the spacecraft computer.

c. Dual Experiment Mounts and Pointing Concept

Figure 7-26 shows the CAOS with both an LST Experiment package and an HEAO experiment capsule mounted thereon. The HEAO is supported by a two-degree of freedom yoke assembly which permits the HEAO sensors to be pointed to any point in the celestial hemisphere opposite to the pointing direction of the LST. The LST is pointed to a target star, and then the Cluster is rolled about the LST center line until the solar array is normal to the Cluster-sunline. This attitude is held while the HEAO is rotated (gear driven) about the LST pointing axis and is tilted in its yoke until the desired HEAO pointing direction is attained. The Cluster is then held in the fine-pointing mode under the control of the LST fine attitude control system while LST and HEAO observations proceed simultaneously. Reconciliation of the pointing programs of the two observatories is required, but that is a minor concession to make to attain the cost-savings that clustering affords.

d. On-Orbit Assembly

The rotation ring assembly (for HEAO support yokes) is attached to the spacecraft structure prior to its placement by the Shuttle into orbit (with the LST attached). Docking cones and probes are installed on the face of the rotation ring to accept mating probes and cones on the yoke assembly.

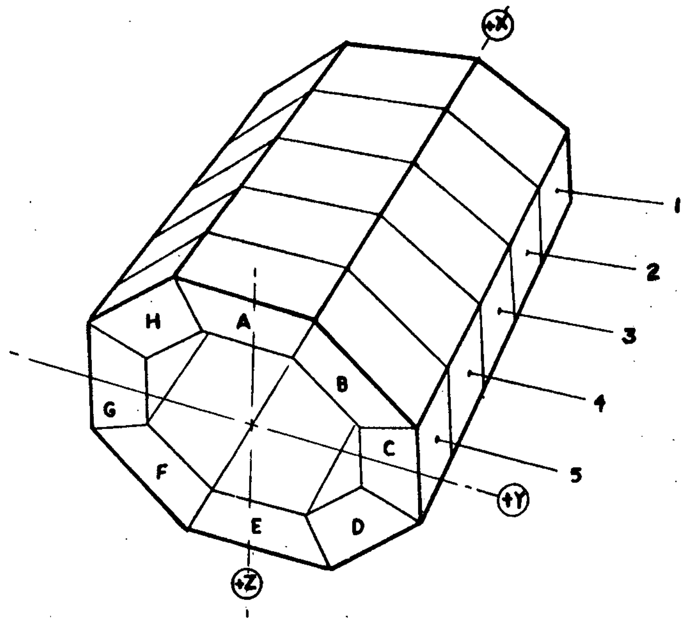
The HEAO is mounted in the yoke assembly on the ground and electrical connections made between the yoke assembly and the HEAO; integral electrical wiring on the yoke connects this interface with a spacecraft umbilical connector on the base of the yoke. The assembly is launched by the Shuttle and docked with the Spacecraft/LST (using Teleoperator, manipulator, or other orbit-assembly assist) to form the Cluster. An umbilical, engaged automatically at the time of docking the yoke assembly to the spacecraft, provides electrical connection of the HEAO to the Spacecraft subsystems.

e. Alternate Cluster Arrangement

The CAOS is also capable of similarly supporting the LSO and LRO payloads simultaneously. The LRO is launched separately and is mounted on the yoke in lieu of the HEAO. The Cluster is stabilized with the LSO pointed to the sun. In order that the LRO may point to any point in the celestial sphere the base of the yoke assembly must be modified to incorporate a hinge and actuation to provide approximately 90° of tilt about an axis parallel to the axis through the LRO support pivots.

f. Location of Modules in Cluster Spacecraft

The locations of the standard modules in the CAOS are shown in Fig. 7-27. The empty compartments identified may be used for the installation of auxiliary sensor modules, designed to be compatible with the standard mechanical and electrical interface provisions of the compartments.



Location	Module	Location	Module
A-1	Secondary Sensing	E-1	Empty
A-2	Empty	E-2	Data Processing
A-3	Empty	E-3	Power Distribution
A-4	Empty	E-4	Empty
A-5	K_u-Band Communication	E-5	S-Band/VHF Communication
B-1	Empty	F-1	Primary Sensing
B-2	Battery Power	F-2	Battery Power
B-3	Reaction Torque	F-3	Reaction Torque
B-4	Battery Power	F-4	Battery Power
B-5	Attitude Control	F-5	Attitude Control
C-1	Empty	G-1	Empty
C-2	Empty	G-2	Empty
C-3	Solar Array Drive	G-3	Solar Array Drive
C-4	Empty	G-4	Empty
C-5	Empty	G-5	Empty
D-1	Primary Sensing	H-1	Empty
D-2	Battery Power	H-2	Battery Power
D-3	Reaction Torque	H-3	Empty
D-4	Battery Power	H-4	Battery Power
D-5	Attitude Control	H-5	Attitude Control

Fig. 7-27 Subsystem Module Locations -
Cluster Astronomical Observatory Spacecraft

C.3

7.5.5.2 Cluster Earth Observatory Spacecraft.

a. General Configuration

The Cluster Earth Observatory Spacecraft (CEOS) shown in Fig. 7-28 is derived from the Standard Earth Observatory Spacecraft (SEOS) described in par. 7.4.2.

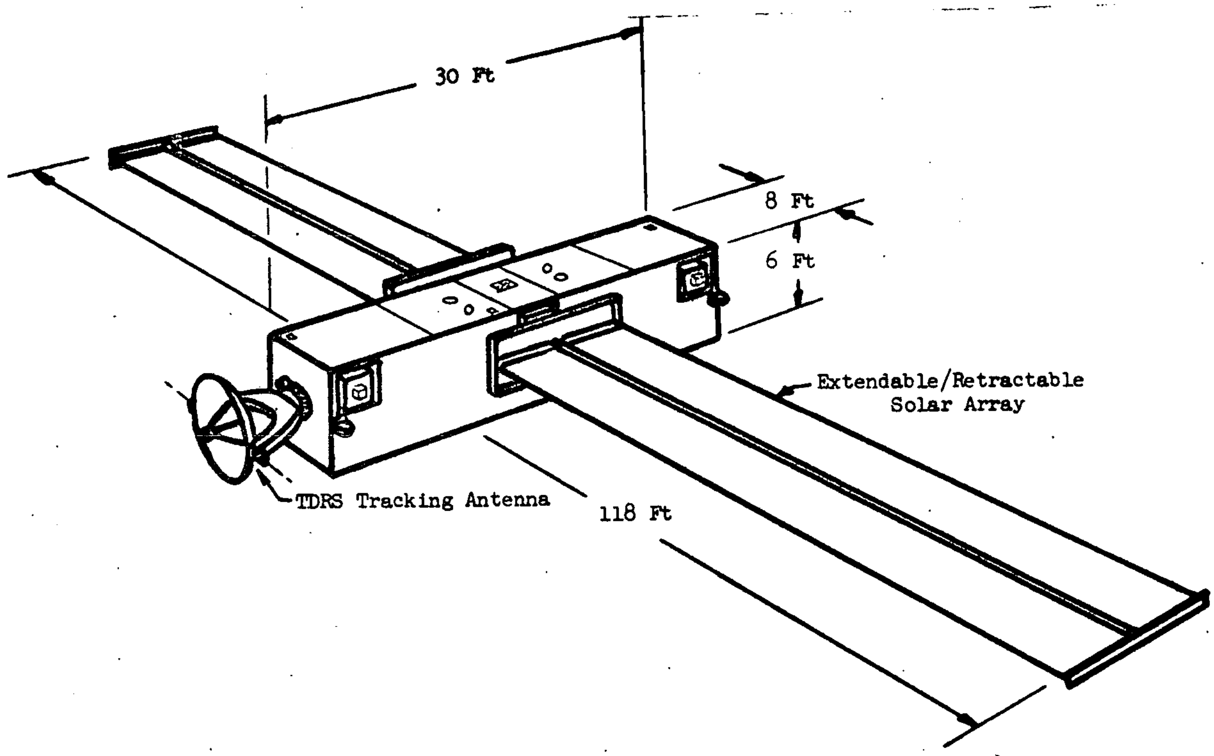


Fig. 7-28 Cluster Earth Observatory Spacecraft

b. Extrapolation of Standard Spacecraft

The principal physical differences between CEOS and SEOS are summarized in the following table:

Physical Characteristic	Standard EOS	Cluster EOS
Length (ft)	20	30
Cross Section (ft)	6 x 8	6 x 8
Subsystem Compartments	18	27
Earth Viewing Surface (ft^2)	160	240
Weight (lbs)	6240	10960
Flexible Solar Array Area (ft^2)	416	1040

c. Electrical Power

The large flexible solar array generates 2600 watts average, the highest estimated power requirement of the CEOS. Eight EPS-6 standard Battery modules are required to maintain the depth of discharge of the NiCd batteries at 15% or less.

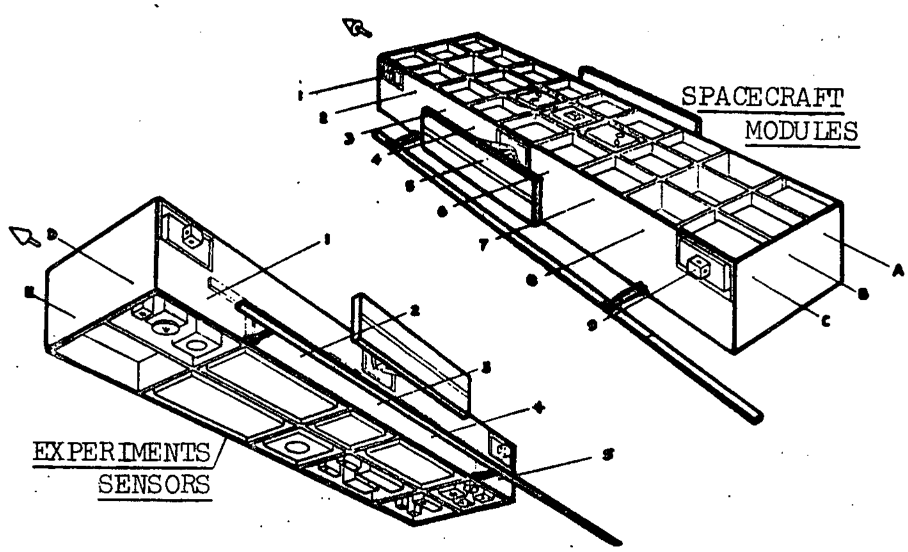
d. S&C and CDPI Subsystem Variants

Other minor increases in the subsystem module complement of the SEOS required to meet the requirements of the CEOS involve adding two modules:

- Module S&C-1 Precision Sensing
- Module CDPI-1 K_u Band Communication

e. Module Locations and Installation

The locations of the standard modules for the Cluster Spacecraft and a representative set of experiments/sensors are shown in Fig. 7-29.



Loc.	Spacecraft Subsystem Module	Loc.	Experiment/Sensor
A-1	Attitude Control	D-1	Surface Composition Mapping Radiometer; Imaging Radiometer; Radar Cloud Top Ranger
A-2	S-Band/VHF Communications		
A-3	Battery Power		
A-4	Battery Power	D-2	Passive Microwave Radiometer (λ = 0.81 cm)
A-5	Solar Power		
A-6	Battery Power	D-3	Synthetic Aperture Radar
A-7	Battery Power	D-4	Passive Microwave Radiometer (λ = 2.81 cm)
A-8	Empty		
A-9	Attitude Control	D-5	Radar Altimeter; Temperature Profile Radiometer
B-1	K_u-Band Communications		
B-2	Empty		Multispectral TV Camera (2)
B-3	Data Processing	E-1	Empty
B-4	Primary Sensing	E-2	Passive Microwave Radiometer (λ = 6.01 cm)
B-5	Secondary Sensing		
B-6	Primary Sensing	E-3	Thematic Mapper
B-7	Power Distribution	E-4	Cloud Physics Radiometer
B-8	Empty		Sea Surface Temperature Radiometer; Passive Microwave Radiometer (λ = 1.67 cm)
B-9	Reaction Torque		
C-1	Attitude Control		
C-2	K_u-Band Communications		Passive Microwave Radiometer (λ = 1.40 cm)
C-3	Battery Power		
C-4	Battery Power	E-5	Ocean Scanning Spectrophotometer
C-5	Solar Power		Atmospheric Pollution Sensor
C-6	Battery Power		Upper Atmosphere Sounder
C-7	Battery Power		
C-8	Empty		
C-9	Attitude Control		

Fig. 7-29 Spacecraft Module and Experiment/Sensor Locations - CEOS

Section 8
SHUTTLE INTERFACES AND SUPPORT EQUIPMENT FOR FUTURE PAYLOADS

For unmanned payloads, the Shuttle is intended to perform the functions of a common carrier, providing delivery services to and from orbit. A wide variety of cargo must be expeditiously handled, both on the ground and in flight; the loading/unloading operations should not tie up the carrier's equipment any longer than necessary. To accomplish this objective, it is necessary to provide well-planned interface equipment and operational methodology.

This section offers a brief over-view of the impact of future low-cost or standardized payloads upon the Shuttle system. A complete dissertation on the payload/Shuttle interface requirements is contained in a separate report, LMSC-D154600, "Preliminary Shuttle/Payload Constraints Analysis."

8.1 General Payload/Shuttle Interfaces

The primary payload interfaces are with: (a) the Shuttle launch base facilities, (b) the Orbiter cargo bay and support subsystems, (c) the Orbiter flight/cargo crew, and (d) the tracking and data relay satellite (TDRS), and/or the ground communications/data network.

The major physical interfaces between unmanned payloads and the Shuttle are:

(a) Adaptors or pallets or cradles
(b) Structural supports and latches
(c) Deployment Mechanisms
(d) Manipulators/Retrieval Devices
(e) Hardline Umbilicals
(f) Payload Service Panels (Electrical, Fluid, Communications, etc.)
(g) Cargo Bay Doors
(h) Payload Deployment Control & Monitor Panels
(i) Payload Test Set

(j) Stowage Provisions for Payload Replacement Modules

(k) Access Provisions to Payloads (Airlock, Hatches, etc.)

(l) Lighting and Visual Access

(m) Purging, Cooling and Contamination Control Provisions

A matrix chart illustrating the interface equipment versus its general payload function is shown in Fig. 8-1 .

The functions and locations of this equipment in the cargo bay must be specified very early in the Shuttle program so that:

(1) compatibility can be planned and designed into each payload

(2) later costly redesigns of Shuttle to accommodate payloads can be avoided

	FUNCTION									
PAYLOAD INTERFACE EQUIPMENT	RENDEZVOUS DOCK, RETRIEVE	STATE VECTOR NAVIG. UPDATE	DATA RECORD RELAY	MONITOR CHECKOUT	CARGO/CREW TRANSFER	ENVIRONMENT CONTROL	FLUID VENT, DUMP	ORBIT DEPLOYMENT	P/L REPAIR, REFURB	INSPECTION
P/L CHECKOUT SET			•	•				•	•	
P/L DISPLAY/CONTROL PANELS	•			•	•			•	•	
P/L ADAPTERS/CRADLES	•							•	•	
P/L MANIPULATION/TELEOPERATORS	•				•				•	
P/L DEPLOYMENT/RETRIEVAL	•				•			•	•	
P/L SPARE MODULE RACK								•	•	
P/L UMBILICALS, SERVICE PANELS		•	•	•		•	•	•	•	
DOCKING EQUIPMENT	•				•				•	
P/L CLOSURE/SHROUD						•				
CARGO CREW PROVISIONS	•			•						•

Fig. 8-1 Types/Functions of Payload Interface Equipment

8.2 Future Payload Interface Equipment Requirements

8.2.1 On-Board Checkout

Based upon historical space hardware failure data, the launch/ascent phase of operations is most likely to cause a fairly large portion of total payload equipment failures. The Shuttle provides an opportunity, either in the transfer orbit or the parking orbit, to assess survival of the launch/ascent phase by verifying the functions of the payload before payload deployment. If a malfunction is found, repairs can be made on orbit or the mission may be aborted and the payload returned to earth.

The Shuttle data bus and computer allocations to payload operations are probably adequate for safety monitoring of the payload, but insufficient for checkout of even one complex payload. (This assumes that payloads may incorporate some degree of built-in test equipment (BITE) but future cost-effective payloads cannot afford the capability for complete self-checkout). Reprogramming the orbiter computer to checkout each different payload is considered to be impractical. An alternative is to utilize a family of "standard" Payload Test Sets (PTS) that can be used during the various phases of payload test and checkout:

(a) at the payload assembly plant for integrated system test,
(b) at the launch base for pre-installation checkout, and
(c) on board the Orbiter.

The Test Set can be located either in the Shuttle cargo bay or in the crew area. The associated controls and displays would be in the crew area. With a dedicated computer, interchangeable program cassettes, and plug-in modules such a PTS could handle multiple payloads and combinations such as a satellite/Tug/Teleoperator.

The payload test set may also be used on revisit missions to verify ascent survival of the spacecraft replacement modules and, after rendezvous and docking, the condition of the retrieved payload satellite as an aid in determing need for repair and adjustment.

8.2.2 Deployment and Retrieval Devices

Deployment devices probably will be Shuttle-supplied, according to current groundrules. Fixed-mounted manipulator arms or front bulkhead-mounted deployment mechanisms were initially considered for the baseline shuttle; a number of other deployment/retrieval devices have since been proposed and are being studied. Among those is a free-flying remote-controlled Teleoperator.

8.2.3 Structural Supports in Cargo Bay

The deployment/retrieval mechanisms are not intended to provide the structural support in the cargo bay for launch/ascent, orbit maneuvering, and entry/landing conditions. Separate support and latch-down hardware must therefore be provided; structural adapters to span between payload and cargo bay structure must be provided also. Additionally, if the payload itself requires a "strongback" support, a cradle structure must be provided on which the payload can be mounted; the cradle would be latched into the cargo bay structure.

8.2.4 Shuttle Flight/Cargo Crew Interfaces

Recent shuttle system descriptions have included, in addition to the baseline Shuttle two-man flight crew, an additional two men to handle, service, and operate payloads, deployment devices, manipulators and other payload interface equipment. These cargo crewmen must be provided with (1) visual and physical access to the cargo bay, (2) the controls and visual aids required for docking to payloads and space stations, and (3) controls for remote operation of manipulators and Teleoperator. They must also have the capability of IVA or EVA to accomplish unscheduled inspections, adjustments, or repairs to payloads and for emergencies. An airlock between the crew compartment and the cargo bay is necessary for crew access to payloads.

8.3 Launch Base Interfaces with Future Payloads

8.3.1 Pre-Installation Operations

Payloads will be checked out and launch-ready before delivery to the Shuttle Maintenance and Checkout Facility (MCF). Here they must be installed in payload adapters or cradles that interface with the load-carrying supports in the cargo bay. Electrical, communications (including data), safety monitoring, and deployment mechanism interfaces, EMI compatibility, and payload cg location should be checked; this probably will require either a mockup or a functional simulator of the shuttle subsystems and equipment involved. The deployment/retrieval mechanisms and the payload test/checkout set (or simulators thereof) should be included in the total assemblage of equipment used in payload pre-installation tests.

8.3.2 Installation of Payload into Cargo Bay

When the payload is installed in the Shuttle cargo bay safety monitoring umbilicals, electrical power/control/communication umbilicals, fluid services (fill, dump, vent or pressurize -- as applicable), and deployment mechanisms/manipulator must be connected, and their operability verified.

8.3.3 Launch Pad Equipment

Most unmanned payloads can be serviced in the MCF and checkout and status monitoring can be accomplished with the Shuttle on-Board Payload Test Set and the Shuttle communications systems. Propulsion stages using storable propellants may be loaded before installation, but payloads and Tugs employing cryogenic propellants require servicing on the launch pad, with the Shuttle in the vertical position. Provisions for purging and conditioning of the cargo bay on the launch pad by Shuttle ground support systems are part of the baseline Shuttle.

8.4 Shuttle Performance Requirements for Future Payloads

8.4.1 Future Payload Weights

"Baseline" payloads as provided in the Mission Model range from 220 to 9700 kg (485 to 21,300 lb). Applying low-cost and refurbishable equipment payload factors, the maximum is estimated to increase to 13,600 kg (30,000 lb) for low inclination LEO payloads, to 3,600 kg (8,000 lb) for sun-synchronous (polar) LEO payloads placed without the use of Tugs, and to 2,250 kg (5,000 lb) for Syneq orbit payloads.

8.4.2 Shuttle Weight Capability for Payloads

Revisit and refurbishment flights require maneuvering capability for rendezvous and docking to LEO satellites. The following requirements for Shuttle performance are the minimum which will be consistent with cost-effective payload systems:

- 29,484 kg (65,000 lb) into a 93 x 185 km, 28,5° transfer orbit with 457 m/s (1,500 fps) of ΔV
- 18,144 kg (40,000 lb) into a 93 x 185 km, 90° transfer orbit, with 335 m/s (1,100 fps) of ΔV
- Capability to provide supplemental ΔV at the expense of payload weight (additional OMS tankage installed in cargo bay)

These requirements are limited to the baseline Shuttle capability; however, were options available, additional capability for the Syneq orbit mission support would be desirable because:

a. Low-Cost Payloads for missions utilizing reusable LO_2/LH_2 Tugs are weight-limited if sized for maximum round trip capability to Syneq orbit.

b. Many LEO-mission payloads that are not individually weight-limited or length limited can be combined on multi-mission Shuttle flights to save transportation costs.

c. Synchronous equatorial and planetary payloads require the use of Tugs carried to LEO on the same Shuttle flight with the payload. The current Shuttle performance limits the weight and size of Tug that can be carried in addition to the payload.

8.4.3 Shuttle Volume Capability

a. Multiple revisits to large observatories in a common orbit plane will require up to 7m (23 ft) of cargo bay length for supplemental OMS propellants and support equipment (such as a Teleoperator) leaving only 11.3m (37 ft) of length for mission payload.

b. Considering multiple payloads, including Tugs, Shuttle capability for a full 4.57m (15 ft) diameter by 18.3m (60 ft) long clear volume for payloads is required.

8.5 Shuttle Subsystem Support for Future Payloads

The Shuttle must supply to payloads mounted in the cargo bay a number of functional services, included among these are:

- Electrical Power
- Orbit Position (Ephemeris and Attitude Data) (for updating payload inertial reference)
- Communications Ground-Link
- Fluid Vent and Dump Provisions

A separate analysis has been made comparing the current baseline Shuttle subsystem capability with the requirements of the various future payloads. In general, the planned Shuttle has adequate capability.

Details of the analysis and quantification of the requirements are contained in the aforementioned separate report, IMSC-D154600.